

Proceedings of the Mars Global Network Mission Workshop

February 6-7, 1990

Francis M. Sturms, Jr.
Workshop Chairman

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GLOBAL NETWORK MISSION WORKSHOP (JPL)
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ABSTRACT

A workshop on the Mars Global Network Mission held at the Jet Propulsion Laboratory (JPL) on February 6 and 7, 1990, was attended by 68 people from JPL, National Aeronautics and Space Administration centers, universities, national laboratories, and industry. Three working sessions on science and exploration objectives, mission and system design concepts, and subsystem technology readiness each addressed three specific questions on implementation concepts for the mission. The workshop generated conclusions for each of the nine questions and also recommended several important science and engineering issues to be studied subsequent to the workshop.

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SECTION 1

INTRODUCTION

1.1 BACKGROUND

These proceedings document a workshop on the Mars Global Network Mission (GNM) held at the Jet Propulsion Laboratory (JPL) on February 6 and 7, 1990.

Mars network missions have been under study for more than a decade. Recently, the GNM has been included in the robotic mission set defined in the National Aeronautics and Space Administration's (NASA's) 90-day study in support of the new Science Exploration Initiative to return humans to the Moon and Mars (Reference 1). As part of the 90-day study, JPL studied the robotic mission set, including a 1998-launched network mission based on penetrator-type landers. The JPL study is documented in Reference 2. The JPL Global Network Mission study team further detailed the penetrator mission in a data package (Reference 3).

Because of the range of possibilities for implementing this type of mission, and because of the new context of the mission as part of the Science Exploration Initiative, a workshop was scheduled to collect ideas about implementation concepts from the science and industrial communities.

1.2 ORGANIZATION OF THE PROCEEDINGS

This introductory section contains background information on the subject workshop. Exhibit 1 is a copy of the workshop invitation letter with attachments stating the purpose and strategy for the workshop. Exhibit 2 is a copy of key material handed out at the beginning of the workshop. This material shows the workshop agenda and the specific issues to be addressed. A copy of Reference 3 was included with the non-JPL invitations.

Section 2 contains an executive summary of the workshop conclusions.

The details of the proceedings are contained in Section 3. Each of the two plenary sessions, the parallel discussion sessions, and the concluding informal session are summarized here. References are made in the parallel session summaries to materials presented or submitted at the workshop. These submittals are contained in the appendixes. Several papers were submitted after the workshop for inclusion in the proceedings. These are also contained in the appendixes.

The proceedings conclude with a list of references and a list of workshop attendees.

January 9, 1990

^F1^
^F2^
^F3^
^F4^
^F5^

Dear Colleague:

You are invited to participate in a workshop on the subject of a Mars Global Network Mission to be held at JPL on February 6 and 7, 1990. This mission has been proposed for a 1998 launch as part of the robotic exploration mission set leading to human exploration of the Moon and Mars.

The workshop has two major objectives: 1) to review and reconfirm the candidate science and exploration mission objectives; 2) to identify implementation options and tradeoffs to meet those objectives. The workshop will help mission planners collect ideas, especially from outside JPL, about applicable unclassified engineering technology and implementation concepts to meet the current science and human exploration initiative objectives for the Mars Global Network Mission.

A copy of the agenda is attached, as well as a list of the specific questions to be addressed at the workshop. All participants are encouraged to select at least one of the session questions of particular interest and to prepare a short brief addressing the issue for discussion at the appropriate session. Bibliography material would also be very useful. These submittals will be published as part of the workshop proceedings.

Also included for your information is a copy of a point design based on an all-penetrator mission that was generated at JPL during the recent Human Exploration Initiative 90-day Study.

A social time is planned for the evening following the first workshop day. Details will be announced at the workshop.

Please respond concerning your attendance at the workshop by Thursday, February 1. Call or write the JPL Global Network Mission study leader, Fran Sturms, at (818) 354-5514, Mail Stop 171-267. Mr. Sturms can also be reached at FTS 792-5514 or through NASAMAIL box FSTURMS.

Exhibit 1. Invitation and Attachments

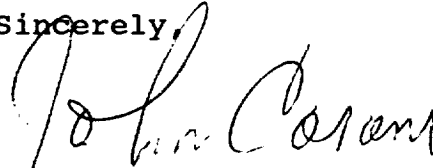
Colleague

- 2 -

January 9, 1990

We at JPL look forward to your participation and hope it will be mutually beneficial as we prepare for this interesting mission.

Sincerely,



John R. Casani
Assistant Laboratory Director
Office of Flight Projects

Attachments

Exhibit 1 (contd)

Mars Global Network Mission Workshop

Dates: Tuesday and Wednesday, 6-7 February 1990

Place: Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91109

Purpose:

To collect ideas, especially from outside JPL, about applicable unclassified engineering technology and implementation concepts to meet the current science and human exploration initiative objectives for a Mars Global Network Mission (GNM).

Agenda:

Day 1	Building 167 Conference Room	
	Plenary Session	
0900	Welcome and Opening Remarks	Casani
0915	Workshop Plan	Sturms
0930	Overview of Science Objectives	Squyres
1000	Human Exploration Objectives	Bell
1030	Break	
1045	Parallel Sessions	
	A - Science and Exploration	
	B - Mission and System Design	
	C - Subsystem Technology	
	(rooms to be announced at plenary session)	
1215	Lunch	
0115	Sessions A,B,C in parallel	
0500	Adjourn	
Day 2	Building 167 Conference Room	
	Plenary Session	
0900	Summary Report, Session A - Science and Exploration	
0930	Summary Report, Session B - Mission and System Design	
1000	Summary Report, Session C - Subsystem Technology	
1030	Break	
1045	Formulation of Workshop Conclusions	
1230	Adjourn formal workshop	
0130	Informal Post-workshop discussions	

Exhibit 1 (contd)

Workshop Strategy:

The primary emphasis of the workshop is on engineering, but with heavy science participation. A review of the science objectives for GNM that were developed at the December meeting of the Mars Science Working Group (MarsSWG) will be presented in the opening plenary session along with the objectives of the Human Exploration Initiative. The heart of the workshop consists of three parallel sessions which will each specifically address three questions concerning implementation of the GNM mission:

Session A - Science and Exploration

- 1) In view of the stated science and human exploration objectives, what are realistic for GNM and what should be allocated to subsequent missions?
- 2) Should there be only one type surface station (e.g. penetrators) or a mix of lander types? Also-how many? where? what lifetime?
- 3) What instruments should be included in the lander and orbiter payloads?

Session B - Mission and System Design

- 1) How do we get to polar sites and is a common lander design feasible for both low latitude and polar sites; both surface and subsurface science?
- 2) What is the best entry system - fixed or deployed aeroshells; parachutes or direct impact?
- 3) What are the desired and achievable accuracies for targeting the landing sites?

Session C - Subsystem Technology

- 1) What technology will help achieve 10-year lifetimes?
- 2) What technology will help survival of high-g landings?
- 3) Are RTGs a workable power subsystem (size, location on the lander)?

A member of the MarsSWG GNM sub-group will attend each parallel session. Each session will have a moderator to maintain the focus on the questions. A session recorder will capture inputs. Participants are encouraged to prepare a short written brief and bibliography material addressing one or more of the session questions to be included in the published proceedings. Position papers will be presented to start off discussion on each question. A workshop consensus will be documented, as well as outstanding issues for further study.

Exhibit 1 (contd)

Session A - Science and Exploration

Room assignment: 264-461B

Moderator: Dan McCleese

Recorder: Matt Golombek

Questions:

- 1) In view of the stated science and human exploration objectives, what are realistic for GNM and what should be allocated to subsequent missions?
- 2) Should there be only one type surface station (e.g. penetrators) or a mix of lander types? Also-how many? where? what lifetime?

Position Statement for questions 1 and 2: Fran Sturms

- 3) What instruments should be included in the lander and orbiter payloads?

Position Statement: Bruce Banerdt

List of attendees:

Bruce Bachofer - GE
Bruce Banerdt - JPL
Don Bickler - JPL
Mike Carr - USGS
Paul Davis - UCLA
Tom Economou - U Chicago
Matt Golombek - JPL
Robert Haberle - ARC
Ron Kahl - JSC
Tony Knight - MMC
Jack Kropp - TRW
Peter Landecker - Hughes
Janet Luhmann - UCLA
Mike Malin - Arizona State
Dan McCleese - JPL
Chris McKay - ARC
David Morrison - ARC
Bruce Murray - Caltech
Dave Smith - JPL
Fran Sturms - JPL
Tomas Svitek - Caltech
Dick Wallace - JPL
Rich Zurek - JPL

Exhibit 2. Key Topics

Session B - Mission and System Design

Room assignment: 167

Moderator: Tom Penn

Recorder: Robert Mostert

Questions:

1) How do we get to polar sites and is a common lander design feasible for both low latitude and polar sites; both surface and subsurface science?

Position statements: Phil Knocke, Jim Burke

2) What is the best entry system - fixed or deployed aeroshells; parachutes or direct impact?

Position statement: Joe Gamble

3) What are the desired and achievable accuracies for targeting the landing sites?

Position statement: Les Sackett

List of attendees:

Arden Albee - Caltech

Norman Alexander - GE

Steven Bailey - JSC

Ed Belbruno - JPL

Dave Bell - JPL

Jim Burke - JPL

Louis Cassel - TRW

Bruce Crandall - Hughes

Manuel Cruz - TRW

Glenn Cunningham - JPL

Alan Friedlander - SAIC

Terry Gamber - MMC

Joe Gamble - JSC

Phil Knocke - JPL

Eric Laurson - Lockheed

Allan Lee - JPL

Bob Mitchell - JPL

Bob Miyake - JPL

Carlos Moreno - JPL

Robert Mostert - JPL

Tom Penn - JPL

Richard Reinert - Ball

Les Sackett - CSDL

Joel Sperans - ARC

Byron Swenson - SAIC

Dan Young - McDAC

Exhibit 2 (contd)

Session C - Subsystem Technology

Room assignment: 238-543

Moderator: Brian Muirhead

Recorder: Bill Nesmith

Questions:

1) What technology will help achieve 10-year lifetimes?

Position statement: Genji Arakaki

2) What technology will help survival of high-g landings?

Position statement: C. Wayne Young

3) Are RTGs a workable power subsystem (size, location on the lander)?

Position statement: Mike Shirbacheh

List of attendees:

Larry Adams - MMC

Genji Arakaki - JPL

Wayne Arens - JPL

John Garvey - McDAC

Paul Gillett - GE

Owen Gwynne - ARC

Robert Karpen - JPL

Brian Muirhead - JPL

Bill Nesmith - JPL

Farley Palmer - Hughes

Dave Ryerson - Sandia

Al Schock - Fairchild

Mike Shirbacheh - JPL

Robert Smolley - TRW

Steve Squyres - Cornell

C. Wayne Young - Sandia

Exhibit 2 (contd)

SECTION 2

SUMMARY OF CONCLUSIONS

2.1 SESSION A: SCIENCE AND EXPLORATION

- 2.1.1 In view of the stated science and human exploration objectives, what are realistic for GNM and what should be allocated to subsequent missions?**

The workshop concluded that most of the stated objectives for this mission should be retained. The meteorology, surface and subsurface chemistry, volatiles, regolith structure, descent imaging, and entry science objectives support both science and exploration. Seismology is of interest primarily to science. No strong arguments have been offered that seismic measurements are needed for exploration purposes. Narrow-band seismology should be retained, but wide-band seismology may have implementation problems on a surface lander, and should be retained only if adequate isolation from lander and surface-generated noise can be accomplished. Postlanding imaging from the lander on the surface can certainly enhance the interpretation of surface measurements, but it is not as important as descent imaging. Orbital objectives, such as aeronomy, support imaging, and other orbital support measurements, are all candidates for implementation by other missions. The workshop strongly recommended that orbital measurements to support surface meteorology be considered. It was also recommended that aeronomy be included on the orbiter if it is possible to do so without seriously complicating the spacecraft carrier design. The Mars Science Working Group (MarsSWG) will convene three small workshops to address questions in the areas of meteorology, seismology, and geochemistry/volatiles/exobiology.

- 2.1.2 Should there be only one type surface station (e.g. penetrators) or a mix of lander types? Also-how many? where? what lifetime?**

A major conclusion of the workshop is that two lander types be used: hard landers for the long-life surface objectives and penetrators for the short-life subsurface objectives. There should be 10 to 20 hard

landers placed relatively evenly at widely separated latitudes (including the polar regions) and longitudes, and at a variety of terrain types. These landers should operate for many years, at least one Mars year at typical 3-sigma spacecraft-design lifetime probabilities and on the order of 10 years at some reduced confidence. Simultaneous data are desired from the long-lived surface landers. Penetrators should be sent to about eight sites, including the polar regions, and need last only a few weeks. Measurements from the penetrators do not need to be simultaneous.

The two types of landers need not be part of the same mission, and they could be launched on different launch vehicles in the same or a different launch opportunity and by different agencies, including those from other nations.

2.1.3 What instruments should be included in the lander and orbiter payloads?

Strawman payloads for each lander type and the orbiter were recommended. All suggestions for the payloads that follow were judged to be preliminary and should be updated in the near future by the MarsSWG workshops.

- (1) Hard lander: high priority; a meteorology station (pressure, temperature, wind, and aerosol measurements at a minimum), narrow-band seismometer, descent imager, alpha-proton-X-ray (a-p-x) instrument, and soil electrochemical analyzer, and instruments for entry measurements of acceleration, pressure, and temperature (probably on the aeroshell). More difficult to accommodate will be a differential scanning calorimeter (DSC), an evolved gas analyzer (EGA) and a neutron spectrometer. A landed imager and impact accelerometer were listed at a lower priority.
- (2) Penetrator forebody: a descent imager, alpha-proton-X-ray instrument, DSC/EGA, neutron spectrometer, impact accelerometer, and gamma-ray spectrometer, and instruments for entry measurements of acceleration, pressure, and temperature (on aeroshell). A soil electrochemical analyzer was listed as a high priority, but it may be difficult to implement without major complexity.

- (3) Penetrator afterbody: soil electrochemical analyzer. Landed imager and alpha-proton-X-ray instrument at lower priority.
- (4) Orbiter: Strong desire for instruments for orbital support measurements, especially for meteorology. An aeronomy package should be investigated, but it has lower priority.

2.2 SESSION B: MISSION AND SYSTEM DESIGN

2.2.1 How do we get to polar sites and is a common lander design OK for both low latitude and polar sites?

The workshop endorsed deployment of all landers from orbit, using an elliptical, polar orbit (see Section 6.2, Submittal No. 1) that can reach all latitudes. There is a 6-month wait for proper lighting for descent imaging, which was judged not to be a big problem. Deployment from hyperbolic approach has many problems and should be considered only for fairly simple penetrators launched on a separate mission. A common design for landers at all latitudes was judged to be possible; the only exception was the design for penetrators that try to penetrate the polar ice cap.

2.2.2 What is the best entry system - fixed or deployed aeroshells; parachutes, or direct impact?

The workshop recommended the use of fixed aeroshells at shallow entry angles as the least risky for launch in 1998. Both deployable aeroshells and the use of steep entry-angle designs show promise, but they have more development schedule risk for a 1998 launch.

Parachutes are recommended and, to reduce g loading, crushable structures for the hard landers and possibly the penetrator afterbodies. The concept of sizing the parachute for the specific altitude target of each lander was offered. Also, a proximity sensor to impart a retro-rocket impulse just before impact could reduce lander impact acceleration from several hundred gs to less than 100 gs .

2.2.3 What are the desired and achievable accuracies for targeting the landing sites?

There is a possible mismatch between desired and achievable landing accuracies. High-probability accuracies are greater than 100 km (3-sigma radius); with additional efforts to control deployment errors, there is some promise of accuracies of 50 to 100 km. Some terrain types desired for targeting may be in the 10- to 50-km range. Achieving this accuracy is questionable without active guidance during atmospheric entry. Target areas should be limited to the 50- to 100-km range. The orbital retromaneuver delta-V should be imparted by a liquid system for greatest accuracy. Guided entry was not judged feasible for these simple landers.

Session B also considered a fourth question concerning alternate mission and lander concepts. The major alternate concepts recommended for further study are (1) spread launches of landers over several opportunities with smaller launch vehicles, (2) separate probe carrier and orbiter communication functions into two missions, (3) attempt a contract for communication services similar to that for Comsat, (4) use a mix of at least two lander types with international partners providing one or more.

2.3 SESSION C: SUBSYSTEM TECHNOLOGY

2.3.1 What technology will help achieve 10-year lifetimes?

The key problem for achieving long lifetimes on the hard landers involves electronics and thermal control. The large number of thermal cycles and electrical on/off cycles (greater than 4000) will stress electronic subsystems beyond the present levels of experience. The failures occur in solder joints and interconnections rather than in the parts themselves. Current electronics fail within 200 to 1000 cycles when thermally cycled from -55 to 100°C. The key to extending the lifetime is in limiting the thermal cycles to a narrower range, e.g., -20 to 20°C. Studies are needed to test the feasibility of thermal-control designs. Testing of electronic designs for large numbers of cycles is important.

2.3.2 What technology will help survival of high-*g* landings?

Current electronic designs have demonstrated survival of the impact accelerations expected for hard landers and penetrators, which range from hundreds to thousands of *gs*. Transverse accelerations may be as high as the axial loads, and rebound accelerations may also be significant. Improved impact models of Mars are needed for design and testing purposes.

Current radioisotope thermoelectric generator (RTG) power-source designs cannot survive the 1000+ *g* impacts that will be experienced on the penetrator afterbody and on an unattenuated hard lander. (See the following question.)

2.3.3 Are RTGs a workable power subsystem (size, location on the lander)?

RTG power sources present a major set of developmental problems for the GNM. Small RTGs of the necessary few watts of power are currently available only for terrestrial use; they can probably be designed to survive a few hundred *gs* and be space and nuclear-safety qualifiable, but they will require considerable design analysis and testing. In the current lander concepts, these lower *g* levels are experienced only in the penetrator forebody. However, temperature control in the forebody is a very severe design problem. Impact attenuation of hard landers and penetrator afterbodies is a possible design approach, but lateral loads may be a problem.

The results from Session A indicate that RTGs may be needed only on the long-life hard landers and not the short-life penetrators. The use of a proximity retro-rocket impulse to lower the lander impact accelerations to less than 100 *gs* could relieve the RTG design problem. Another way to lower the RTG impact loads is separation of the RTG from the lander prior to impact, using either the descent parachute or a separate chute. This would require an umbilical for power transfer and analysis of other problems, such as lateral loads, landing orientation, and temperature control.

SECTION 3

DETAILED PROCEEDINGS

3.1 PLENARY SESSION 1

The first plenary session began with a welcome by John Casani, JPL's Assistant Laboratory Director for Flight Projects. The GNM study leader, Fran Sturms, then reviewed the mechanics of the workshop as presented in the invitation and workshop handout materials (see paragraph 1.2).

Jim Martin brought a challenge from NASA code E to "be innovative."

The next three presentations at the opening session served to review the science and exploration objectives for GNM. These objectives were not to be viewed as firm requirements, but rather as goals to serve as guidelines for the subsequent discussions. The viewgraphs presented during this part of the workshop are referenced in this section as figures. (In some cases, handwritten slides have been typed for these proceedings.)

The first presenter was Mike Carr, chairman of code E's Mars Science Working Group (MarsSWG). Dr. Carr put the mission objectives into the context of the series of robotic missions leading to the first human landings on Mars. GNM completes the reconnaissance or global assessment phase of Mars exploration (Figure 1). Primarily, the GNM provides ground truth at a number of sites on Mars. The missions following GNM comprise the validation phase of Mars exploration; this phase increases confidence of our understanding of Mars to levels required to evaluate and select sites and to design and fly missions to land humans (Figure 2). He pointed out that this mission has been of interest to the science community for many years and showed a recent set of objectives from a code-EL workshop (Figure 3).

The next presenter was Steve Squyres, chairman of the GNM subgroup of MarsSWG. Dr. Squyres presented the science objectives for GNM in six parts: atmospheric science, internal structure, geochemistry and mineralogy, volatiles, surface morphology, and regolith structure. (See Figures 4 through 9.) Each of these experiment areas were detailed as to the type of measurements, the kind of instrumentation, and the

mission requirements. Additional science desired involves soil oxidation state, aeronomy, magnetometry, heat flow, and entry science (Figure 10). The science objectives divide naturally into short-lived subsurface science and long-lived surface science. This in turn implies two types of surface landers: short-lived penetrators and long-lived hard-surface landers. (See Figure 11.) Finally, it was pointed out that both types of landers will desire targeting to terrain types as small as 50 km, at latitudes and longitudes over the whole planet, and at a wide range of surface altitudes.

The third presenter was Dave Smith, representing the Science Exploration Initiative (SEI) Mission Analysis and System Engineering (MASE) organization at Johnson Space Center (JSC). He pointed out that the robotic missions obtain both science and engineering data to support the mission, spacecraft, and equipment design for the human missions (Figure 12). The SEI objectives overlap the pure science objectives in many areas. The key SEI activities to be supported by additional knowledge about Mars are aeromaneuvers in the Mars atmosphere and site selection for the human landings (Figures 13 and 14). Specific objectives for GNM include supporting the site selection for the subsequent robotic sample return mission and validating the global resource maps of Mars compiled from remote sensing on such missions as Mars Observer (Figure 15).

3.2 SESSION A: SCIENCE AND EXPLORATION

Session A on Science and Exploration was held on February 6. Papers submitted for discussion are reproduced in Section 6.1 and referenced here by numbers in parentheses.

Dan McCleese (moderator) presented an overview of some of the issues to be discussed (1). These included science objectives for the GNM, relevance of the GNM to the SEI, the role of the orbiter, and the implementation approach.

Fran Sturms presented a position paper on the GNM mission design (2). The objectives fit into four basic categories of long- and short-lived surface landers and long- and short-lived subsurface penetrators. Fran proposed surface hard landers only for the GNM, delaying subsurface objectives for later in the mission set. The scientists present overwhelmingly opposed such a solution, insisting that subsurface

volatile and chemistry information is crucial for calibrating the remote sensing data obtained by Mars Observer.

A broad-ranging discussion followed that questioned the need for large Titan launches for the GNM: Perhaps a greater number of smaller launches would be more appropriate. This seemed more in keeping with the request for alternative strategies and concepts for implementing the GNM and for involving international partners in the mission.

A philosophical discussion followed that ranged from the SEI interests for the GNM to a simpler, smaller, and more fiscally conservative mission than the one generally considered. Dave Morrison stressed that the GNM should focus on objectives unique to a network mission (3): the simultaneity of observations for meteorology and seismology, and the ability to go places that will not be visited by the other robotic exploration missions. Bruce Murray discussed at length his views on what the GNM should encompass. He thought that a simple, hard-lander mission involving launch on small expendable vehicles is, politically, the most sensible approach. These landers would measure properties for which enough information already exists to enable design of a useful experiment. This would entail measuring atmospheric properties and surface imaging. This view submits that not enough is known about the seismology of Mars to allow meaningful measurements. Most of the other scientists vehemently disagreed with this position: Seismological measurements remain the only way to determine fundamental properties of the interior. It was also pointed out, however, that there is no fundamental incompatibility between meteorology and seismic measurements at the same long-lived surface station, provided there is no long-term mast or boom that would wave in a wind.

After lunch, Mike Carr discussed the science objectives (4) attributed to the GNM by the MarsSWG. After a discussion that lasted most of the afternoon, it was concluded that these science objectives remain the best for the GNM. A table of the needs for the GNM for a variety of disciplines was presented (4). This table shows that about 20 surface stations are needed for meteorology, 10 to 20 are needed for seismology, and 10 or fewer are required for volatile and mineralogy/chemistry science. A possible break between long-lived surface stations and short-term penetrators was iterated from MarsSWG discussions.

The group overwhelmingly supported plans for a number of small science workshops that focus on such critical questions as how much does a simple surface station that is not firmly anchored to the

ground degrade the seismic measurements? Another is how many meteorology stations are needed and what additional measurements from orbit need to be made simultaneously? A third is how many landers are needed for geochemistry and volatile science, and are subsurface measurements required? These questions will be addressed by three upcoming workshops on seismology, meteorology, and geochemistry/volatiles/exobiology objectives for the GNM.

Bruce Murray then presented a concept for adding balloons to the hard landers now being considered for the GNM (5). These balloons would allow high-resolution imaging of the Martian surface as well as meteorology in the boundary layer. Tethered-, anchored-, and snake-balloon concepts were presented and recommended for consideration as a payload option.

Dan McCleese directed the focus of discussion to issues useful for mission designers. A variety of questions and suggestions surfaced. They included: What is the current engineering design for a hard lander? Is a hybrid hard lander/penetrator in which a hard lander has a spike for effective coupling to the ground possible? Does a hard lander perform a scientifically satisfactory group of measurements?

Dave Morrison suggested a strawman payload for a hard lander (6). The payload would include instruments for meteorology, seismology, surface chemistry, soil physics, and imaging. A second class of vehicle was proposed to accompany the hard landers, namely short-lived penetrators that concentrated on the volatile, mineralogy, geochemistry science goals. Steve Squyres summed up this apparent division between science on penetrators and hard landers (7). This implementation strategy looked promising to the group and more design work on it was recommended.

To focus on a possible strawman payload for this type of mission, Bruce Banerdt presented a compilation of science instruments and their masses, power, and data-rate requirements (8). Outside of discussion about some of the instruments (e.g., a neutron spectrometer vs a gamma-ray spectrometer), this strawman was considered reasonable, although most believed it was still a little early in the mission-design activities to accept a strawman payload. It was agreed that the mass, power, and data-rate estimates need to be evaluated by the workshops that focus on meteorology, seismology, and geochemistry/volatiles/exobiology.

Janet Luhmann presented an excellent summary of what was known about the aeronomy of Mars and the need for additional measurements (9). Most agreed that the orbiter will offer a good opportunity for an aeronomy payload. The package proposed is under 100 kg, although some of the instruments require unimpeded ram directions (without landers in the way) as well as fairly long booms. The spacecraft orientation required to provide this ram direction may conflict with the communications function of the orbiter, although a separate communications satellite was suggested to alleviate this mismatch.

Bob Haberle suggested that atmospheric measurements from orbit will be needed at the time the surface meteorology measurements are made. While the science from these measurements is quite compatible with the major goals of a network mission, most present were unwilling to choose the orbiting atmospheric measurements over the aeronomy measurements, given that an aeronomy mission had previously been given a high scientific priority. Everyone agreed, however, that a balance has to be achieved between lander and orbiter science and between orbiter and surface meteorology, although this balance has not yet been reached.

Tom Economou presented information on a small, rugged alpha-proton-X-ray instrument being developed. Although most scientists saw the utility of this type of instrument, difficulties in interpreting the results were noted, given that this instrument samples only a thin layer. Anyone interested in more information is encouraged to talk with Tom.

The meeting ended with a wrap-up session that summarized what would be presented at the plenary session.

Subsequent to the workshop, Paul Davis submitted a paper on the objectives of a Mars seismology experiment (10).

3.3 SESSION B: MISSION AND SYSTEM DESIGN

Session B of the Global Network Mission Workshop focused on issues relevant to the overall mission and system design. These issues included the causes for the network mission to exceed reasonable costs and feasibility and the conceptual design of the 90-day study penetrators. Tom Penn was designated the moderator of this session, and Robert Mostert the recorder. Materials submitted for discussion are reproduced in Section 6.2 and referenced here by numbers in parentheses.

The issues raised in this session included (1) the ability to place surface landers at the polar regions, (2) the feasibility of one long-lived common lander design for both high- and low-latitude sites, (3) the possibility of one common lander design to do both surface and subsurface science, (4) the most feasible system to enter the Martian atmosphere—fixed vs deployable aeroshells—and the best terminal decelerator—parachutes vs direct impact, and (5) the desired and achievable accuracies for targeting the landing sites.

The first issue raised was emplacement of landers at the polar regions. Could a long-lived common lander design be used? Could surface and subsurface objectives be achieved with a single long-lived common lander design? Phil Knocke and Jim Burke presented opening position papers to facilitate discussion.

A mission design proposing an elliptical polar orbit to place landers at the higher latitudes was presented by Phil Knocke (1). His premise is based on a requirement to emplace landers above 80 degrees latitude. Also, good lighting angles are necessary for descent imaging. These two requirements constitute the major mission-design drivers. A previous site-synchronous orbit design was able to emplace landers at the higher latitudes upon approach, but at very steep entry angles and poor lighting conditions. The elliptical polar orbit allows landers to be placed anywhere on the Martian surface at reasonable lighting conditions and at shallower entry angles. The landers would be deployed from orbit instead of on approach. This helps to avoid the larger landing dispersion of approach-deployed landers. The orbit allows a second pass over the planet, which gives the mission an element of redundancy for lander emplacement. As such, this orbit design would permit aeronomy experiments while the spacecraft waits to deploy the landers.

Jim Burke spoke about system design considerations and options (2). This entailed the Mars network mission objectives, approach, and desired results. He stated that the goals of the mission need to be prioritized. Once the priorities have been established, different design options could be eliminated. A systems design tree that shows the different options of the different stages of emplacing a lander on the surface of Mars from atmospheric entry to surface landing was shown. It serves to remind the designer of the different choices available. For example, a designer is reminded that the lander could use a parachute or a ballute for atmospheric terminal descent. Jim showed a couple of lander designs done in the past (e.g., Viking) using the design tree to point out the alternative options chosen versus those options eliminated.

Alan Friedlander of SAIC presented global contour maps of the available communications time per sol between a lander and the orbiter summed over all orbiter passes during that sol (3). The maps help to clarify the one-sol communication interval between a lander at any given place on Mars and the orbiter, using the elliptical polar orbit. Alan also showed a configuration of a spacecraft with stacked aeroshells at launch (3). After orbit insertion, each aeroshell would be individually deployed from the spacecraft bus. Each aeroshell, containing two penetrators, would then be its own spacecraft on its way to Mars. At Mars, the aeroshells would enter the atmosphere individually towards their designated landing sites. Alan's proposal is based on a communications infrastructure that would be already available at Mars. A short discussion ensued about the communications industry providing the necessary infrastructure for a set fee. Friedlander also contributed an update to his previous study of lander success probabilities (5), but this was not discussed.

Manny Cruz of TRW presented the results of studies relevant to steep entry flight path angles (6). The 90-day study penetrator is the baseline lander. The study is based on the use of ballistic coefficients to assess the terminal velocity, ground track, and angle of attack of the lander with respect to steep entry flight-path angles. The results show that different ballistic coefficients do not adversely affect the ground track, but landing accuracies decrease with shallower entry flight-path angles. An aeroshell or its equivalent—designed with any ballistic coefficient—that enters the atmosphere at steep flight path angles will have a good landing accuracy. The terminal velocity increases and the angle of attack decreases with larger (>10) ballistic coefficients. The disadvantage of entering the atmosphere at steep entry flight path angles seems to be the greater temperatures the entry vehicle will encounter.

A short-lived penetrator for the Martian poles derived from the Comet Rendezvous Asteroid Flyby (CRAF) penetrator was presented by Rich Reinert of Ball Aerospace (7). He suggested that a Martian polar penetrator (a CRAF penetrator derivative) as an individual mission would not be difficult. A modification would not be complicated because the science objectives and the instrument complement for the Martian polar caps are similar to those for the CRAF mission. The proposal assumes that the polar caps are of composition similar to that of a comet. Jim Martin asked if the CRAF penetrator could be used at lower latitudes; Rich responded by suggesting that a major reconfiguration would be necessary.

A general discussion followed that summarized and concluded the first set of issues. It was agreed that the polar caps could be reached by landers with a good mission design. A single, long-lived common lander could probably not be designed for both the higher and lower latitudes. A cost estimate would be helpful to confirm this conclusion. The issue of placing surface and subsurface science instruments on a single lander was not addressed at this point, nor was the issue of alternative types of landers (i.e., hard vs soft). Jim Martin suggested that, with international partners, a mix of landers might be possible. A suggestion was also made to use different opportunities to send different or a different number of landers to Mars. This would allow designers to ensure mission success in the case of a failure, to affect designs as needed, and to spread the loss of a single mission failure over several missions. Martin also suggested a look at the 1996 opportunity for a quick, inexpensive mission.

The second key issue raised at this session was the question, "Can deployable aeroshell technology be used for this mission or should fixed aeroshells be used?" Also, the question of using parachutes for terminal descent versus direct impact of the lander was raised. Joe Gamble of Johnson Space Center gave an opening position paper to begin the discussion.

Joe Gamble presented results of a study that would help identify deployment dispersion problems upon Martian atmospheric entry (8). (Note: This contribution was rewritten and resubmitted after the workshop.) The 90-day-study penetrator baseline design was used to determine a ballistic coefficient. He compared the deployment of aeroshells from orbit with that from approach and the dispersion problems that result due to terminal velocities. His results show that deployment from approach is risky unless large entry angles are used with very large g loads. He also observed that parachute deployment altitude will be critical in establishing terminal velocities. The higher the altitude, the greater the mach number upon descent, which implies higher g impacts.

Considerations and issues in an entry- and terminal-descent-system design were prepared by Byron Swenson of SAIC, but presented by Alan Friedlander (9). The presentation showed different types of entry-system configurations done in the past. Byron Swenson has been studying deployable aeroshells, but did not use one for the 90-day study, because he did not think it could be developed by 1998. He has been looking at different ways to jettison landers from aeroshells. Yet, a deployable aeroshell is worth consideration. It would ease the packaging problem at

launch. Rich Reinert mentioned that Massachusetts Institute of Technology (MIT) has been looking at deployable accelerators, and MIT should be contacted to identify progress.

Arden Albee of Caltech showed lander design drivers from the standpoint of packaging at launch and deployment from the spacecraft versus each lander's mass and structure or entry system, such as an aeroshell (10). He used a design tree to show the results of different landers that have been built or studied. They are the Soviet Mars 3, the Soviet Venera 8, the Viking Lander, and the Ames penetrator of 1977. He showed the choices made by each mission to highlight the different design choices that can be considered in placing landers on Mars. He discussed other lander designs that have been built and that are not penetrators. Some aspects of a penetrator have not been thoroughly studied, such as a long-lived power source that would survive high-g impacts. Alan Friedlander commented that the instrument payload is very restricted on a penetrator. In addition, Arden is concerned with the wide variation in altitudes at which landers will be placed without taking into consideration the entry angle and the resulting terminal velocity at that altitude.

The discussion that followed summarized the issue and concluded that because fixed aeroshells are well known and the technology is available, the landers will be kept simple. Deployable aeroshells will have to be excluded from any 1996 opportunity because of the development schedule that will be required. However, they may not necessarily need to be ruled out for a 1998 opportunity. The issue of a parachute versus direct impact was raised because of the landing-site altitudes. At what altitude is a parachute to be deployed for a high-altitude landing site in order to decelerate the lander to an acceptable velocity? Jim Martin mentioned that for the Viking lander, project management is concerned with any altitude above 2 km, because the Viking landers would not have decelerated to an acceptable velocity. Retro-rockets or retro proximity fuses, like those the Soviets use, were suggested. Hence, a study is needed to identify the altitudes at which the landers can be placed with certain decelerators. A range of altitude regions will need to be studied, in addition to the height-range variability.

The last key issue raised in this session was that of desired and achievable accuracies for targeting surface landing sites. Les Sackett of The Charles Stark Draper Laboratory (CSDL) was asked to present an opening position paper (11).

The desired landing accuracy for a long-lived surface lander with meteorology and seismology instruments is on the order of 100 km. The desired accuracies for short-lived landers with chemistry and mineralogy instruments are on the order of tens of kilometers. Les Sackett of CSDL, using the 90-day study penetrator as a baseline, showed that a penetrator deployed from approach and deployed from orbit at a flight path angle of -20 deg would have an approximate landing accuracy of 300 to 350 km. A penetrator deployed at a flight path angle of -15 deg would have an approximate landing accuracy of 450 to 500 km. Again, steeper flight path angles improve landing accuracies. A comparison of deployment from approach and deployment from orbit suggest that the landing accuracy dispersion is not so different.

The ensuing discussion centered on how accurately landing sites could be targeted and with how much complexity and cost. It is clear that targeting control systems can be added to landers to decrease landing accuracy errors, but these subsystems would increase the complexity, costs, and mass of each lander. Hence, decisions must be made to clearly identify the desired landing accuracies.

The remainder of the time was used to present and discuss alternative issues and design considerations that did not seem pertinent to the issues already discussed.

Bruce Crandall of Hughes Aircraft Co. told the session that Hughes has been working on very high-*g* subsystems to be used for Space Defense Initiative (SDI) (12). Hence, he could not speak at length. Significant and mature high-*g* technologies exist in the areas of electronics, propulsion, imaging, and guidance systems. Moreover, he raised a launch-vehicle issue. Is a Titan IV/Centaur the launch vehicle that this mission will be permitted to use? The use of that launch vehicle implies a number one priority launch, and that has not been proclaimed for this mission. He suggested using two or more Atlas/Centaur launch vehicles. The total would have equivalent mass capabilities to that of the Titan IV/Centaur and would provide multiple launches, which could satisfy any redundancy or probability of a mission success requirement. Each bus would carry fewer landers, but this strategy would decrease the chances of failing to emplace all the landers. That is, if all 20 landers were on one spacecraft, and the first one could not successfully deploy—which is required for deployment of the following landers—the mission would be a complete failure. Having but a few landers per bus and a total of four spacecraft would overcome this potential problem. Finally, if only 10 landers were placed on two separate spacecraft for redundancy, the Titan

IV/Centaur probably could not be launched from the same launch pad within a 20-day window. Atlas/Centaurs could be turned around within 20 days.

Carlos Moreno of JPL showed a conceptual configuration of a hard lander on which he has been working (13). It is based on a JPL study done in 1988 and is similar to that used by the Soviets in the early 1970s. A parachute would be used for terminal descent, and a crushable impact absorber would be used to withstand the final surface impact. The lander uses an RTG as its power source and includes meteorology, seismology, and soil-oxidant instruments in addition to an alpha-proton-X-ray spectrometer. An attempt to add descent imaging will be made on this design. The design drivers for this lander include the size and mass of the RTG, whether or not it can survive high-*g* impacts, and the size and mass of the memory necessary for data storage. The latter is also a function of data rate and transmission time. However, some think that memory will not be a driver for this lander. The present design mass is approximately 10 to 15 kg.

Jim Burke of JPL presented a tethered balloon option for a payload on a lander. (This is the Session A contribution No. 5 by Murray.) It is an alternative to descent imaging. It would also be used for boundary layer measurements. Bruce Murray of Caltech has been working on this concept with some of his graduate students. However, several problems are associated with a tethered balloon. First, packaging a pressurized tank with gas is difficult. Second, filling the balloon with gas is not trivial. Third, a good anchor is necessary to overcome any wind problems. A solution to the anchor is the use of some sort of snake that is dragged over the surface as the balloon pulls it. Another issue is the imaging done by the balloon. This imaging will not permit identification of the lander location. Jim Burke will continue to work with Bruce Murray to evaluate tethered balloons.

John Garvey of McDonnell Douglas (MDAC) spoke about the capabilities of a Delta launch vehicle using two or more Delta vehicles to do a network mission (14). This presentation was similar to that of Bruce Crandall with respect to launch vehicles. More than one Delta vehicle can be launched in a 20-day launch window.

Other issues discussed following these presentations included instrument payload feasibility. Which technologies can reach maturity for this mission? Subsurface chemical-analysis instruments would be a severe design driver. Jim Martin is not sure that we are looking at one

mission to be done in 1998 only or if we are looking at other opportunities. A concept of more than one network mission would have merit for the human exploration group, which would be able to help select certain landing sites.

The conclusions and recommendations of this session are summarized in the workshop summary section.

Steve Bailey submitted a paper after the workshop for inclusion in the proceedings (15).

3.4 SESSION C: SUBSYSTEM TECHNOLOGY

Session C on Subsystem Technology was held on February 6. Materials submitted for discussion are reproduced in Section 6.3 and are referenced here by numbers in parentheses.

In the opening position statement on 10-year lifetime survival, Genji Arakaki pointed out that the big concern is the large number (greater than 4000) of temperature cycles, both from the Mars day-night cycle and from the electrical power on/off switching. Technology options include new packaging, coatings, and expansion boards. Brian Muirhead pointed out that the subsurface components on penetrators might see smaller fluctuations. However, the polar sites are very cold (150 K) and the soil thermal conductivity is uncertain. Genji Arakaki stated that current electronic packaging technology is good for 1000 to 2000 cycles at -55 to 100°C. Testing of existing packaging methods for over 4000 cycles needs to be done. Byron Swenson remarked that the long lifetime follows a high-g landing, which may add to the problem. Wayne Young recommended looking at technology used in the 20-year life for nuclear weapons storage. Current programs are looking at high-g designs: CRAF penetrator, tested to 400 gs, and Smart Pebbles, tested to 48,000 gs.

After lunch, Wayne Young presented information on the Sandia penetrator experience (1). Sandia has a lot of experience with penetrator technology and many tools are available for use in the GNM program. Young concludes that technology exists to develop the Mars penetrators.

Dave Ryerson described the Sandia Telemetry Department practices and rules for building high-shock instrument electronics (2). A key observation was that shock attenuators and energy absorbers have not

worked well in Sandia's experience because of amplification of *gs* at some frequencies and rebound loads.

Tom Komakek described high-*g* design concerns for RF hardware (3).

Farley Palmer presented information concerning the GNM technology questions from Hughes' experience base (4).

Mike Shirbacheh presented material provided by Teledyne Energy Systems on the use of small RTGs in terrestrial applications (5).

Al Schock presented design concepts for penetrator RTGs based on extensions of existing Fairchild RTG work (6).

3.5 PLENARY SESSION 2

The second plenary session of the workshop convened at 9 a.m. on February 7. The purpose of this session was to hear summary reports from the parallel sessions of the previous day and to develop workshop conclusions. The viewgraphs presented during this part of the workshop are referenced as figures in this section. (In some cases, handwritten slides have been typed for these proceedings.)

The first presenter was Steve Squyres, representing the Session A moderator, Dan McCleese. (See Figures 16 through 21.) It was recommended that GNM focus on simultaneous global measurements in a wide variety of terrain types and in particular in those types that would not be visited by subsequent missions (e.g., the high latitudes and terrain too rugged for the safe landing of rovers and humans). Boundary-layer measurements were added to the meteorology objectives. The pros and cons of the two main lander types were discussed, as well as some variants, such as hard landers with "spikes" to penetrate the soil. A number of science and engineering issues were identified. The three science issues will be taken up by special MarsSWG working groups.

Tom Penn, the Session B moderator, presented summary results for each of the three questions addressed in the Mission and System Design session (Figures 22 and 23). In response to a fourth question (Figure 24), a system-design tree (Figure 25) was presented to show trades for alternate concepts. Several recommendations were made

to study additional mission and system designs as funding will allow (Figures 26 and 27).

Brian Muirhead, Session C moderator, presented a summary of technology status and issues, and recommendations for each of the three questions addressed in the Subsystem Technology session (Figures 28 through 31). There is no clear need for new technology; however, considerable advanced development is needed in most areas for high-*g* penetrators (Figure 32). An innovative option recommended for study is the use of microthrusters to reduce landing accelerations to tens of *gs*. To be ready for a 1998 mission, the top three areas needing immediate advanced development are deployable aeroshells; small, high-impact RTGs; and small retropropulsion for softer landings.

Fran Sturms led a discussion of the workshop conclusions, using six forms (Figures 33 through 38) with blanks to be filled in. These completed forms, along with notes and comments recorded to capture details not apparent on the somewhat simplified forms, were used to develop the summary conclusions documented in Section 2.

3.6 INFORMAL SESSION

The workshop was formally adjourned at 12:30 p.m. on February 7, 1990, at the end of the second plenary session. After lunch, an informal session was convened to allow those able to stay to have additional discussions on items of interest.

A number of ideas on how to obtain the desired imaging of the lander sites were discussed. It was pointed out that the large data storage and recovery necessary in descent imaging could be avoided by taking images from tethered balloons subsequent to landing or from cameras lofted by mortar from the surface. These approaches would also get the desired atmospheric boundary-layer measurements. However, these techniques involve pointing-control problems. It was the consensus that descent imaging was the best way to get images, which could be continuously nested from orbital resolution levels down to submeter resolution at the site. The case for postlanding imaging was judged to be weak, especially if the descent images were good enough to identify surface effects on the meteorology measurements. Landed imaging would, however, allow identification of postlanding weather effects such as frosts or wind deposits of dust around the lander site. Relatively unobstructed imaging is desired.

A number of ideas for getting the seismometer away from the potentially disturbing influence of the surface meteorology station were discussed. The seismometer could be deployed by arms away from the lander where it could be screwed or driven into the soil or otherwise be made to have good surface contact. The arm could also release the seismometer, leaving a data and power umbilical. Another technique would fire the seismometer away by means of a mortar.

The advantages of deployable aeroshells were discussed further. The main advantages are smaller mass and less attachment space on the orbiter. Testing deployable aeroshells in a Mars-like environment will be difficult, and this contributes to the development schedule risk for a 1998 launch.

Methods for "stacking" fixed aeroshells so that they take up less room on the orbiter were discussed. Two ideas mentioned were a sideward "frisbee" type release (which also imparts the desired spin) and ejection from a stack in a "rifled" launch tube. The former may offer the advantage of selection of specific aeroshells that have been tailored to specific types of landing sites, e.g., with parachutes sized for different altitudes.

DISTRIBUTION OF MATERIALS ENVIRONMENTAL CONDITIONS GEOLOGICAL, BIOLOGICAL, AND CLIMATOLOGICAL HISTORY

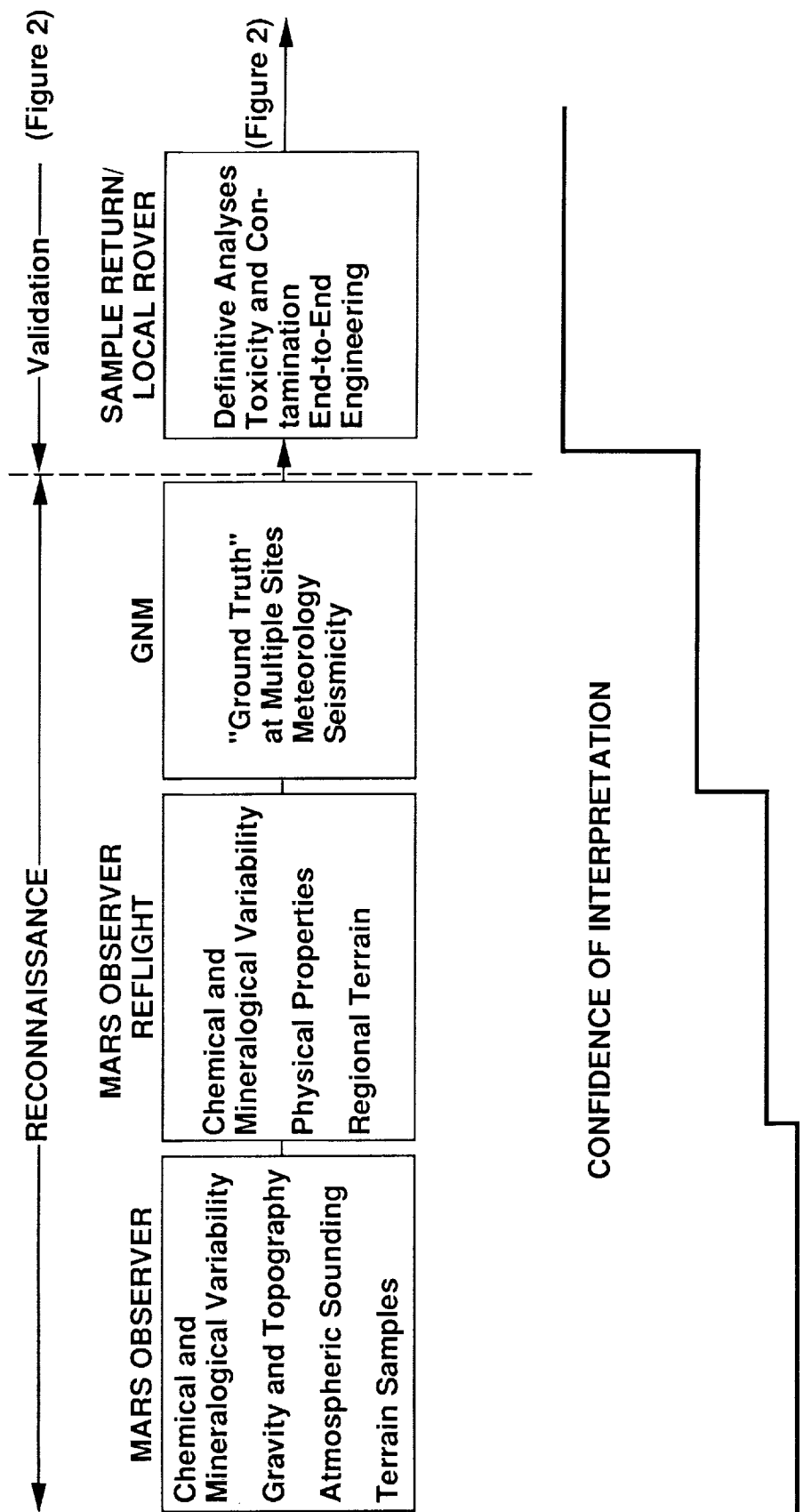


Figure 1. Phase I: Global Assessment

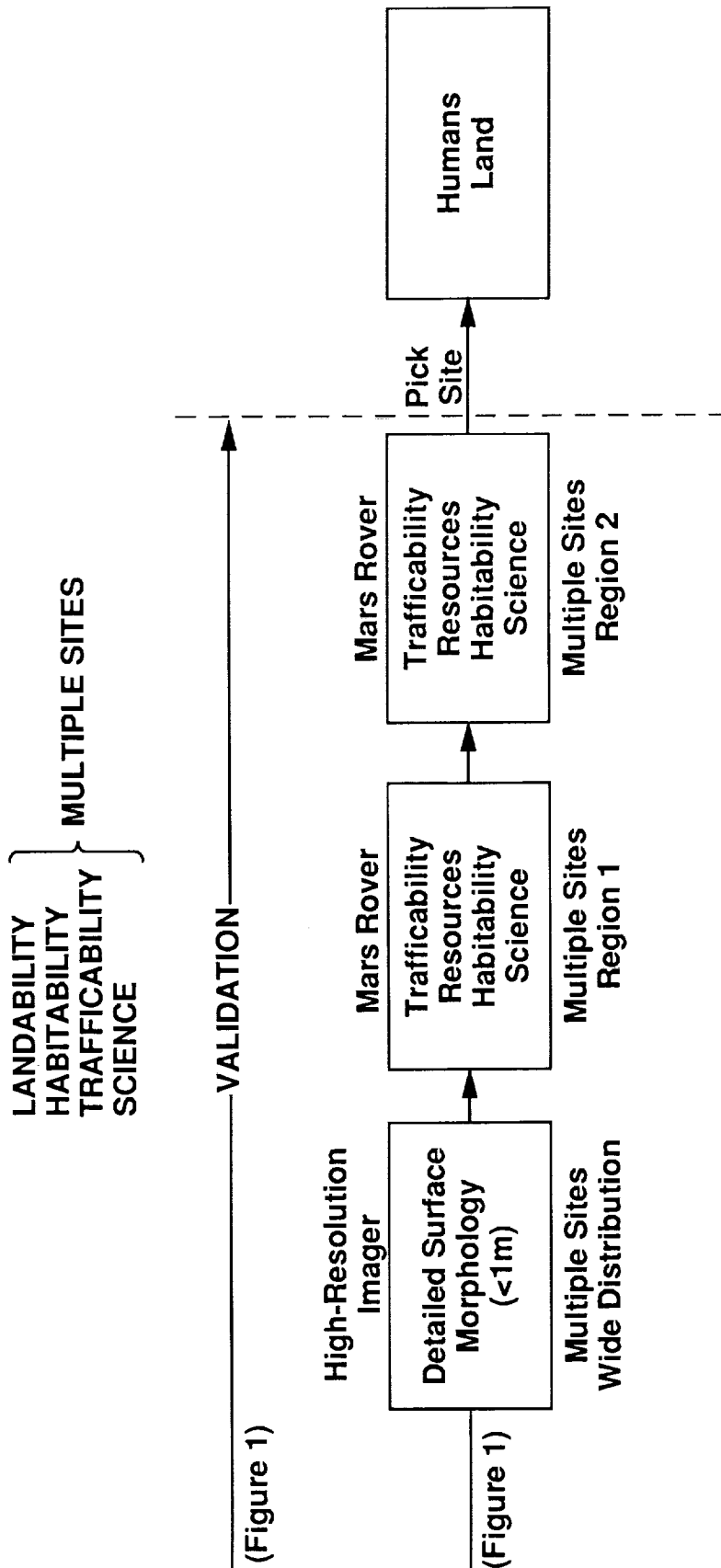


Figure 2. Phase II: Site Evaluation



Global Network Mission

(Two Launches with 12 Penetrators Per Launch)



- Detailed Study of Globally Distributed Sites
- Establish Long-Term Seismic & Meteorological Network
- Program Significance
 - Early Detailed Characterization of Surface and Subsurface Properties
 - Measure Global Distribution of Water
 - Long-Term Meteorological Observations
 - Global Seismicity
 - Hi-Resolution Imaging of Localized Sites
 - Establish Navigation Aids for Subsequent Landings
 - Provide Multiple Atmospheric Profiles

Figure 3. Global Network Mission

SCIENCE OBJECTIVES: ATMOSPHERIC SCIENCE

- SURFACE PRESSURE (SEASONAL VARIATIONS, WEATHER SYSTEMS, WAVES, ATMOSPHERIC TIDES)
- AIR TEMPERATURE AND WINDS (BOUNDARY LAYER, WEATHER SYSTEMS, DUST-RAISING MECHANISMS)
- AEROSOL PROPERTIES (DUST LOADING, RADIATIVE PROPERTIES)
- HUMIDITY (WATER VAPOR TRANSPORT, SURFACE/ATMOSPHERE EXCHANGE)

INSTRUMENTATION: METEOROLOGY PACKAGE

- T, P, WIND SENSORS
- SKY RADIOMETER
- P₂O₅ HYGROMETER?

MISSION REQUIREMENTS:

- SURFACE PLACEMENT REQUIRED
- LONG LIFE REQUIRED
- MANY (≥ 20) STATIONS REQUIRED
- GLOBAL PLACEMENT REQUIRED, INCLUDING POLAR REGIONS (MINIMAL LATITUDE/LONGITUDE RESTRICTIONS)

Figure 4. Science Objectives: Atmospheric Science

SCIENCE OBJECTIVES: INTERNAL STRUCTURE

- MARTIAN SEISMICITY AND RELATION TO GEOLOGY
- SEISMIC ATTENUATION PROPERTIES
- PRESENCE, SIZE, AND PHYSICAL STATE OF CORE
- THICKNESS, VERTICAL STRUCTURE, AND LATERAL STRUCTURE OF CRUST

INSTRUMENTATION: SEISMOMETER

- SEISMOMETER TECHNOLOGY IS WELL ADVANCED
- CAGING/RELEASE MECHANISM REQUIRED
- INTRINSIC DATA RATE IS VERY HIGH; EVENT RECOGNITION SYSTEM IS REQUIRED

MISSION REQUIREMENTS:

- SUBSURFACE PLACEMENT PREFERRED, BUT SURFACE PLACEMENT MAY BE ADEQUATE FOR MOST OBJECTIVES
- LONG LIFE REQUIRED
- MANY (≥ 10) STATIONS REQUIRED
- GLOBAL PLACEMENT REQUIRED (MINIMAL LATITUDE/LONGITUDE RESTRICTIONS)
- ACCURATE TARGETING REQUIRED FOR LOCAL NETWORK(S)

Figure 5. Science Objectives: Internal Structure

SCIENCE OBJECTIVES: GEOCHEMISTRY AND MINERALOGY

- MAJOR ELEMENT GEOCHEMISTRY (ROCK TYPE AND PETROGENESIS)
- SELECTED TRACE ELEMENT CONCENTRATIONS AND RATIOS (e.g., NATURAL RADIONUCLIDES)
- MINERALOGY (PHASE ASSEMBLAGES, HYDROUS MINERALS, CARBONATES)
- SAMPLE WIDE RANGE OF ROCK TYPES
 - ANCIENT CRATERED TERRAIN
 - INTERMEDIATE AND YOUNG VOLCANICS
 - POSSIBLE SILICIC VOLCANICS
 - POSSIBLE AQUEOUS SEDIMENTS/CARBONATES

INSTRUMENTATION: GRS, α -P-X, DSC/EGA

- GAMMA-RAY SPECTROMETER (GRS)
 - DETECTS MAJOR ELEMENTS AND NATURAL RADIONUCLIDES
 - SAMPLES LARGE VOLUME WITH NO SAMPLING MECHANISM
 - AT DEPTH, DETECTION OF ALL BUT K/U/TH REQUIRES NON RADIOISOTOPIC NEUTRON SOURCE
 - GE REQUIRES $T \leq 120$ K, NaI HAS POOR RESOLUTION
 - MATERIALS INTERFERENCE IS A PROBLEM
 - CANNOT BE USED WITH RTG NEARBY

Figure 6. Science Objectives: Geochemistry and Mineralogy

GEOCHEMISTRY AND MINERALOGY (continued)

- **α -P-X**
 - DETECTS MAJOR ELEMENTS
 - NO COOLING, INTERFERENCE, OR RTG PROBLEMS
 - REQUIRES DOOR OR SAMPLING MECHANISM
 - SAMPLES VERY THIN LAYER
- **DSC/EGA**
 - DOES SOME MINERALOGY; ESPECIALLY GOOD WITH HYDROUS MINERALS AND CARBONATES
 - REQUIRES SAMPLING MECHANISM
 - COMPLEX INSTRUMENT

MISSION REQUIREMENTS:

- **SUBSURFACE SAMPLING REQUIRED**
- **SHORT LIFE ADEQUATE**
- **ACCURATE TARGETING REQUIRED**

Figure 6. (contd)

SCIENCE OBJECTIVES: VOLATILES

- POLAR LAYERED DEPOSITS COMPOSITION
- CONCENTRATION AND DEPTH OF GROUND ICE
- HYDROUS MINERALS, CARBONATES, ETC.

INSTRUMENTATION: DSC/EGA, N-SPECTROMETER

- N-SPECTROMETER DETERMINES H CONCENTRATION
- SAMPLES LARGE VOLUME WITH NO SAMPLING MECHANISM
- REQUIRES NEUTRON SOURCE, RTG, RADIOISOTOPE, 14 MeV PULSED NEUTRON GENERATOR

MISSION REQUIREMENTS:

- SUBSURFACE SAMPLING REQUIRED
- SHORT LIFE ADEQUATE
- HIGH LATITUDE REQUIRED

Figure 7. Science Objectives: Volatiles

SCIENCE OBJECTIVES: SURFACE MORPHOLOGY

- HIGH-RESOLUTION CHARACTERIZATION OF LOCAL SURFACE SITES FOR A VARIETY OF GEOLOGIC INVESTIGATIONS
- NESTED IN LOWER-RESOLUTION COVERAGE FOR CONTEXTUAL INFORMATION
- IMPORTANT FOR INTERPRETATION OF GEOCHEMICAL RESULTS

INSTRUMENTATION: DESCENT IMAGER

- COMPACT OPTICS/CCD SYSTEMS ARE RELATIVELY STRAIGHTFORWARD
- VERY HIGH DATA RATE IMPLIES SUBSTANTIAL MEMORY REQUIREMENTS

MISSION REQUIREMENTS:

- INDEPENDENT OF SURFACE/SUBSURFACE OPERATION
- INDEPENDENT OF LONG/SHORT LIFE

Figure 8. Science Objectives: Surface Morphology

SCIENCE OBJECTIVES: REGOLITH STRUCTURE

- REGOLITH STRENGTH PROPERTIES VS. DEPTH
- ANCILLARY DEPTH INFORMATION

INSTRUMENTATION: ACCELEROMETER

- SMALL, SIMPLE, RUGGED
- HIGH INSTANTANEOUS DATA RATE

MISSION REQUIREMENTS:

- SUBSURFACE SAMPLING REQUIRED
- SHORT LIFE ADEQUATE

Figure 9. Science Objectives: Regolith Structure

NOTES

OTHER POSSIBLE SCIENCE

- SOIL OXIDATION STATE
- AERONOMY PACKAGE ON ORBITER
- MAGNETOMETRY
- HEAT FLOW
- ENTRY SCIENCE

**GEOLOGIC UNIT AND LATITUDE REQUIREMENTS
ALSO IMPLY ELEVATION REQUIREMENTS**

**APPARENT NATURAL DIVISION INTO
SHORT-LIVED/SUBSURFACE SCIENCE AND
LONG-LIVED/SURFACE SCIENCE**

Figure 10. Additional Science

TWO VEHICLE TYPES

A) ~8 SHORT-LIVED PENETRATORS

- GRS (NaI; Ge POLAR?)
- α -P-X
- N-SPECTROMETER/14 MeV SOURCE
- DSC/EGA
- ACCELEROMETER
- DESCENT IMAGING

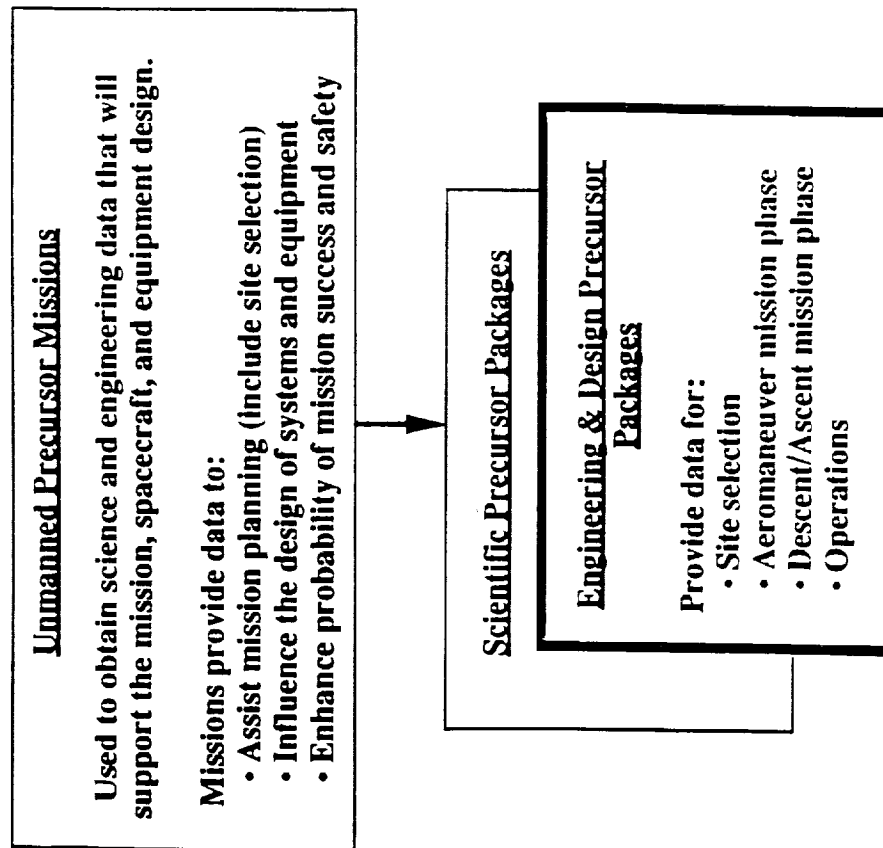
B) ~16 LONG-LIVED HARD LANDERS

- SEISMOMETER
- METEOROLOGY PACKAGE
- DESCENT IMAGING
- α -P-X

**BOTH REQUIRE ACCURATE TARGETING
AND MINIMAL LATITUDE/LONGITUDE
RESTRICTIONS**

Figure 11. Two Vehicle Types

Mars Precursor Mission Requirements



Note: In some cases an overlap of desired or needed data will occur between scientific and engineering precursor missions.

Figure 12. Mars Precursor Mission Requirements

Mars Engineering & Design Precursor Mission Requirements

Aeromaneuver Mission Phase

- Upper atmospheric characterization down to 25km altitude; from actual aerocaptures replicating manned mission approach conditions
 - Average properties
 - Periodic variations
 - Range of unpredictable variations ("pothole" variability)
- Orbital beacon - not needed but desirable. Any transponder that can give accurate range and range rate in any Mars orbit will do

Candidate Site Selection

- Engineering precursors needed to validate safety and select actual landing points within candidate sites
 - Surface slope variations
 - Surface bearing strength
 - Potential landing hazards

Descent/Ascent Phase

- Precursor requirements mirror those needed for site selection

Operations (communications and surface operations)

- No precursor missions needed

Figure 13. Mars Engineering and Design Precursor Mission Requirements

Precursor Requirement - Conclusion

Precursor priority summary

- Atmospheric properties precursor mission critical for aerobrake design
- Surface site safety
 - Slope
 - Bearing strength
 - Obstacles

Figure 14. Precursor Requirement -- Conclusion

SOME OBJECTIVES FOR GLOBAL NETWORK MISSION:

- **SITE SELECTION FOR 2001 SAMPLE RETURN**
 - **1 METER RESOLUTION MAP (HORIZONTAL & VERTICAL) OVER 2-3 Km.**
- **SITE ASSESSMENT**
- **GLOBAL RESOURCES?**
 - **MAP @ 100 Km RESOLUTION?**
 - **SUBSURFACE WATER?**

Figure 15. Some MASE Objectives for GNM

MARS GLOBAL NETWORK MISSION WORKSHOP

FEBRUARY 6-7, 1990

SCIENCE AND EXPLORATION

SUMMARY REPORT

DAN MCCLEESE, MODERATOR

MATT GOLOMBEK, RECORDER

Figure 16. Session A: Science and Exploration

SCIENCE OBJECTIVES

- WHAT ARE VALID SCIENCE OBJECTIVES FOR THIS MISSION?
- OBJECTIVES INVOLVING:
 - MEASUREMENTS IN A WIDE VARIETY OF GEOLOGIC TERRAINS
 - MEASUREMENTS IN LOCATIONS THAT CANNOT BE REACHED BY OTHER MEANS (E.G., TOO HIGH LATITUDE, TOO RUGGED, ETC.)
 - SIMULTANEOUS GLOBAL OBSERVATIONS
- GLOBAL SEISMIC NETWORK
 - DEEP INTERNAL STRUCTURE
 - CRUSTAL STRUCTURE
 - SEISMIC ACTIVITY
- GLOBAL METEOROLOGICAL NETWORK
 - GLOBAL CIRCULATION
 - DUST AND WATER TRANSPORT
 - BOUNDARY LAYER

Figure 17. Science Objectives

SCIENCE OBJECTIVES (continued)

- **VOLATILES**
 - **GLOBAL INVENTORY/DISTRIBUTION**
 - **POLAR DEPOSITS**
 - **RESOURCES**
- **GEOCHEMISTRY/MINERALOGY**
 - **ROCK TYPES/PHASE ASSEMBLAGES/PETROGENESIS**
 - **GROUND TRUTH FOR REMOTE SENSING**
 - **EMPHASIS ON BEDROCK**
- **SURFACE MORPHOLOGY**
 - **GEOLOGIC STUDIES**
 - **HAZARD ASSESSMENT**
 - **CONTEXT FOR OTHER MEASUREMENTS**
- **REGOLITH STRUCTURE**

Figure 17. (contd)

MISSION DESIGN OPTIONS

- LONG-LIVED HARD LANDERS ONLY
 - METEOROLOGY
 - LIMITED SEISMOLOGY
 - LIMITED GEOCHEMISTRY/VOLATILES
- SHORT-LIVED PENETRATORS ONLY
 - GEOCHEMISTRY/VOLATILES
- LONG-LIVED PENETRATORS ONLY
 - METEOROLOGY
 - SEISMOLOGY
 - GEOCHEMISTRY/VOLATILES
- LONG-LIVED HARD LANDERS AND SHORT-LIVED PENETRATORS
 - METEOROLOGY
 - LIMITED SEISMOLOGY
 - GEOCHEMISTRY/VOLATILES
- LONG-LIVED "PENETRATOR/LANDER"
 - METEOROLOGY
 - IMPROVED SEISMOLOGY
 - IMPROVED GEOCHEMISTRY/VOLATILES

ALL OF THESE ALLOW DESCENT IMAGING
- "BUOYANT KITE"
 - IMAGING
 - BOUNDARY LAYER STUDIES

Figure 18. Mission Design Options

PROS & CONS

- PENETRATORS - PRO:
 - ENABLE DEEP SAMPLING
 - SOME HERITAGE (MILITARY, CRAF)
- PENETRATORS - CON:
 - SEVERE INSTRUMENT ACCOMMODATION CONSTRAINTS
 - HIGH G'S REQUIRED
- HARD LANDERS - PRO:
 - SIMPLER INSTRUMENT ACCOMMODATION
 - LOWER G'S POSSIBLE
- HARD LANDERS - CON:
 - DEEP SAMPLING VERY DIFFICULT
 - LITTLE HERITAGE
- TWO VEHICLE TYPES - PRO:
 - ACHIEVES MORE SCIENCE WITH SIMPLER SYSTEMS THAN OTHER APPROACHES
 - OFFERS PROGRAMMATIC FLEXIBILITY
- TWO VEHICLE TYPES - CON:
 - REQUIRES TWO INDEPENDENT AND PARALLEL DEVELOPMENT EFFORTS
 - PROGRAMMATIC FLEXIBILITY IS A TWO-EDGED SWORD

Figure 19. Pros and Cons of the Options

ISSUES & ACTION ITEMS - SCIENCE

- SEISMOLOGY
 - HOW SEVERELY IS SEISMIC SCIENCE DEGRADED BY SURFACE PLACEMENT? BY A "PENETRATOR/LANDER"?
 - HOW MANY SEISMIC STATIONS ARE REALLY NECESSARY?
- METEOROLOGY
 - HOW MANY METEOROLOGY STATIONS ARE REALLY NECESSARY?
 - WHAT IS THE TRADEOFF OF LANDED VS. ORBITAL METEOROLOGY?
- GEOCHEMISTRY/VOLATILES
 - HOW SEVERELY IS GEOCHEMISTRY/VOLATILES SCIENCE DEGRADED BY SURFACE PLACEMENT? BY A "PENETRATOR/LANDER"?
 - WHAT ARE THE REAL REQUIREMENTS FOR PLACEMENT CAPABILITIES (ACCURACY, ELEVATION)?

Figure 20. Issues and Action Items: Science

ISSUES & ACTION ITEMS - ENGINEERING

- **WHAT ARE VIABLE HARD LANDER AND "PENETRATOR/LANDER" DESIGNS?**
- **CAN RTG'S BE ACCOMMODATED IN PENETRATORS? (THERMAL, G-LOADS)**
- **WHAT LANDING ACCURACY IS FEASIBLE?**
- **WHAT IS THE MAXIMUM LANDING ELEVATION FEASIBLE?**
- **HOW FEASIBLE IS ACCOMMODATION OF THE FULL MARS AERONOMY PACKAGE ON THE ORBITER?**
- **HOW NECESSARY IS MARS AERONOMY FOR PLANNING AND EXECUTION OF AEROCAPTURE?**

Figure 21. Issues and Action Items: Engineering

MARS GLOBAL NETWORK MISSION WORKSHOP

FEBRUARY 6 –7, 1990

MISSION AND SYSTEM DESIGN

SUMMARY REPORT

TOM PENN, MODERATOR

ROBERT MOSTERT, RECORDER

Figure 22. Session B: Mission and System Design

SESSION B

MISSION AND SYSTEM DESIGN

QUESTION 1:

HOW DO WE GET TO POLAR SITES, AND IS A COMMON LANDER DESIGN FEASIBLE FOR BOTH LOW LATITUDE AND POLAR SITES, AND FOR BOTH SURFACE AND SUBSURFACE SCIENCE?

POSITIONS:

PHIL KNOCKE (ORBIT DESIGN) AND JIM BURKE (LANDER OPTIONS)

RESULTS:

**A) CAN WE REACH THE POLES? YES WITH KNOCKE'S ORBIT DESIGN
YES FROM APPROACH; OPPORTUNITY
DEPENDENT**

B) COMMON DESIGN FOR POLAR/LOW LATITUDE?

**PENETRATORS: PROBABLY NO
OTHER TYPES: TBD**

C) COMMON DESIGN FOR SURFACE/SUBSURFACE SCIENCE?

- **WE EMBRACED CONCEPT OF LONG LIFE/SURFACE SCIENCE
VS. SHORT LIFE/SUBSURFACE SCIENCE**

OTHER:

**CONSIDER CRAFT PENETRATOR AS POLAR (ICE/SNOW) PENETRATORS
(DOES POLAR MEAN ICE/SNOW OR DUNE FIELDS?)**

Figure 23. Assigned Questions

SESSION B

MISSION AND SYSTEM DESIGN (CONT'D)

QUESTION 2:

WHAT IS BEST ENTRY SYSTEM: FIXED OR DEPLOYABLE AEROSHELLS; PARACHUTES OR DIRECT IMPACT?

POSITION:

JOE GAMBLE (AEROSHELLS)

RESULTS:

FIXED AEROSHELLS:

- CAN COMMIT TO THEM WITHOUT RISK
- GREATLY CONSTRAINS THE DESIGN

DEPLOYABLES:

- SHOW GREAT PROMISE
- MUST COMMIT TO TECH DEVELOPMENT
- NOT READY FOR '96 OPPORTUNITY

PARACHUTES VS DIRECT IMPACT:

**USE DEPLOYABLE AERODYNAMIC DECELERATOR
NOT A CRITICAL ISSUE**

OTHER:

ACHIEVABLE ALTITUDES FOR LANDING SITES:

- 2-3 PARACHUTE SIZES ON LANDERS
- VARIABLE PENETRATION DEPTH, ETC.
- DESIGN MUST HANDLE WIDE ALTITUDE RANGE (-2/+6 km?)
- PUSHES FOR MULTIPLE SYSTEMS TO SOME EXTENT

Figure 23. (contd)

SESSION B

MISSION AND SYSTEM DESIGN (CONT'D)

QUESTION 3:

WHAT ARE DESIRED AND ACHIEVABLE ACCURACIES FOR TARGETING LANDERS?

DESIRED: LONG LIFE/MET/SEIS: 100 km RADIUS
 SHORT LIFE/CHEM/MIN: 10's km RADIUS

- ALSO HIGH ENTRY ANGLE (STEEP VS. SHALLOW ENTRY)
FROM ORBIT OR APPROACH

ACHIEVABLE:

<u>COMPLEXITY/COST</u>	<u>PERFORMANCE (3σ)</u>	<u>IMPLEMENTATION</u>
LOW	200 km Dn Range 1 σ 50 km X Range 1 σ	<ul style="list-style-type: none">- NO IMPULSE CONTROL- SPINNER/SOLID MOTOR- NO FPA ERROR CONTROL
MED	40 km RADIUS	<ul style="list-style-type: none">- BALLISTIC ENTRY- CONTROLLED IMPULSE- CONTROLLED FPA ERROR- VIKING TYPE
HIGH	10 km RADIUS 1 σ	<ul style="list-style-type: none">- PERFECT IMPULSE EXECUTION- COMPENSATE FOR P,T VARIATION- LEAVES ONLY NAV ERRORS- REMOVE WIND AND MAP ERRORS
HIGH ²	FEW km	

Figure 23. (contd)

SESSION B

MISSION AND SYSTEM DESIGN (CONT'D)

QUESTION 4: IDENTIFY ALTERNATE MISSION AND LANDER CONCEPTS

- BRUCE CRANDALL, HUGHES: LV'S/PROBE CARRIERS
- CARLOS MORENO, JPL: CLAMSHELLS
- JIM BURKE, JPL: BALLOONS
- JOHN GARVEY, MDAC: DELTA II/PROBE CARRIERS

SEE FIGURE 25

Figure 24. Alternate Concepts Using System-Design Tree

SYSTEMS DESIGN TREE

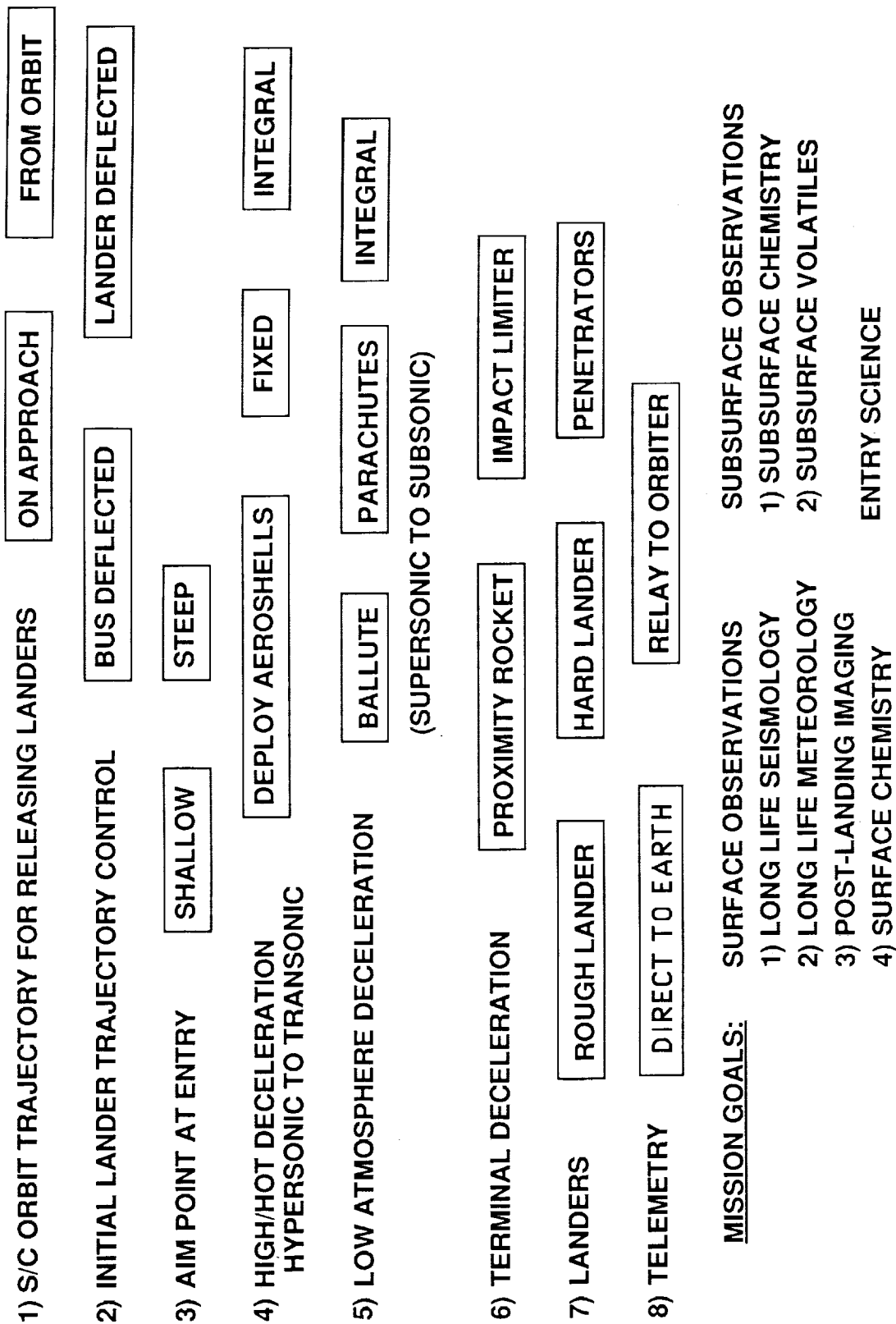


Figure 25. System-Design Tree

SYSTEMS DESIGN TREE (continued)

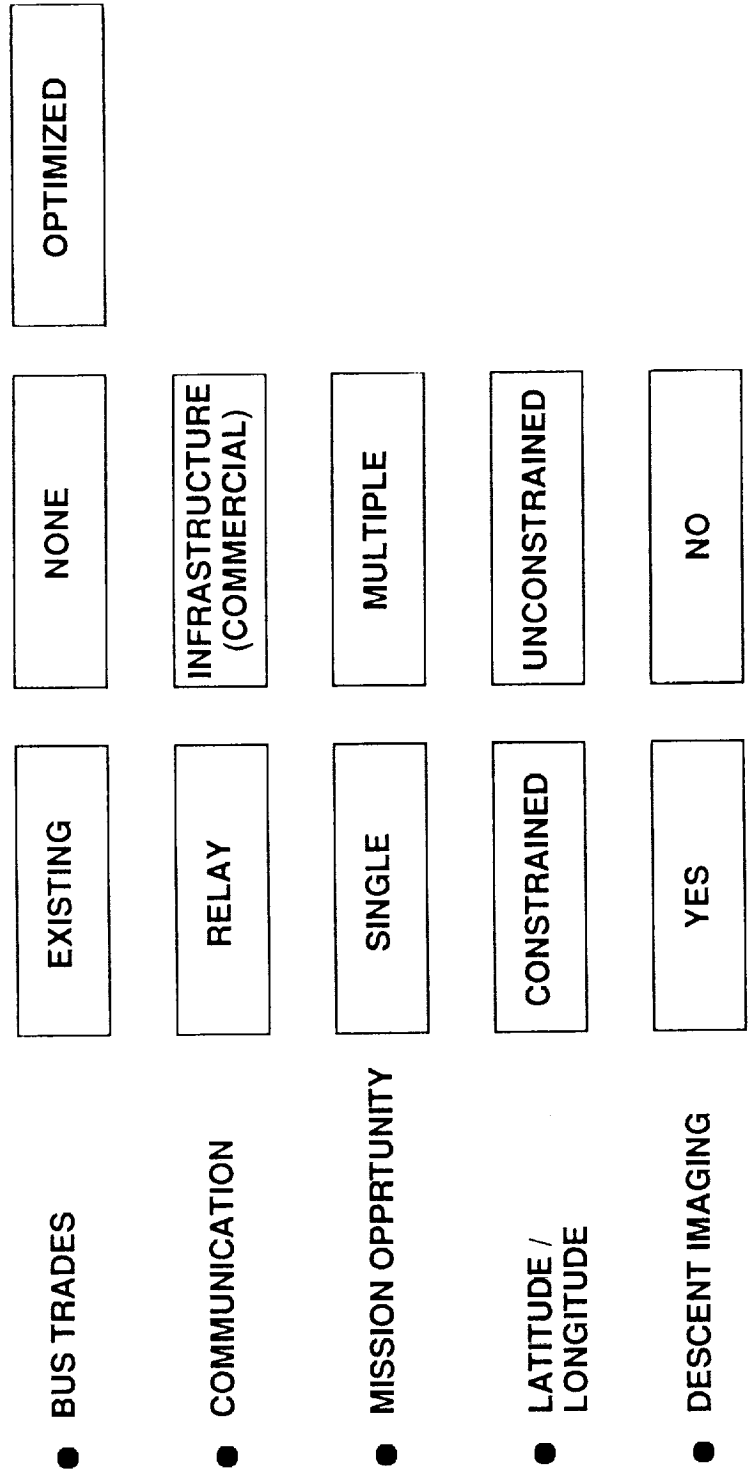


Figure 25. (contd)

SESSION B

MISSION RECOMMENDATIONS

- **CONSIDER SPREADING PENETRATORS OVER SEVERAL LAUNCH OPPORTUNITIES**
- **CONSIDER LAUNCHING PROBE CARRIERS AND COMM/RELAY INFRASTRUCTURE INDEPENDENTLY**
- **CONSIDER INNOVATIVE COMM INFRASTRUCTURE CONCEPTS:**
 - **COMMERICAL CONTRACT TO PROVIDE MARS COMMUNICATION SERVICES**
 - **CONTRACT FOR SERVICE RATHER THAN S/C/LV/UPPER STAGE/OPS**
 - **AWARDS AND INCENTIVE FEES SUCH AS THOSE FOR EARTH-BASED COMSATS**
 - **ALLOW MANY OPPORTUNITIES FOR INTERNATIONAL PARTICIPATION/COOPERATION**
 - **MAKE INFRASTRUCTURE CONSISTENT WITH COMMERCIAL LAUNCH - SERVICE METHODOLOGY**

Figure 26. Mission Recommendations

SESSION B

SYSTEM DESIGN RECOMMENDATIONS

- RECOMMEND A MIX OF LANDER TYPES (AT LEAST TWO)
- OPPORTUNITY TO OPTIMIZE DESIGN OF EACH LANDER TO FIT ITS FUNCTION; THIS COULD SAVE MONEY OVER A COMPLEX MULTI-FUNCTION LANDER
- INTERNATIONAL COOPERATION OPPORTUNITIES
- ABILITY TO HANDLE A RANGE OF ENVIRONMENTS
- DILEMMA

SESSION RECOMMENDED MAINTAINING OPTIONS
VS.
LIMITED FUNDING/SCOPE OF GNM STUDY

Figure 27. System Design Recommendations

Mars Global Network Mission Workshop

February 6-7, 1990

Subsystem Technology Session

Summary Report

Brian Muirhead, Moderator

Bill Nesmith, Recorder

Figure 28. Session C: Summary Report: Subsystem Technology

QUESTION: What technology will help achieve 10-year lifetimes?

ASSUMPTIONS:

- Limited power is available for temperature control and wide temperature swings may occur
- >4000 thermal and/or electrical on/off cycles may occur for various subsystems/components (e.g., telecom)

TECHNOLOGY STATUS/ISSUES:

- Current electronics fail by thermal cycling from -55 to 100°C at 200 to 1000 cycles
- Limitations could exist in solder joints and interconnections. Parts not thought to be a problem
- Batteries require narrow range of temperature control (e.g., +/- 20°C)
- Ni-Cad batteries demonstrated for 10K discharge/charge cycles
- Li batteries striving for 1000 discharge/charge cycles
- Contamination by dust may be a problem, especially after a high impact

CONCLUSIONS/RECOMMENDATIONS

- Current technology may be able to meet lifetime and cycle environment requirements, but actual design temperature range may have to be controlled to a narrow band (e.g., 0 +/- 20°C).
- Immediate design studies are needed to determine if a feasible temperature-control design exists
- Test representative electronic designs for their lifetimes, and pursue advanced technology development to correct any problems
- New technology options exist to deal with electronic packaging problems, but they may be high risk within a limited schedule

Figure 29. Question 1

QUESTION: What technology will help survival of high-g landings?

ASSUMPTIONS:

- g-levels will be in 100s to 1000s of *gs* depending on velocity and characteristics of the material impacted, except for retro-controlled landings
- Transverse levels can be as high as axial levels and are dependent on hard-to-control factors such as impact angle
- Current JPL flight experience indicates that packaged electronics can survive 3000 *gs* at 2000-3000 Hz

TECHNOLOGY STATUS/ISSUES:

- Sandia has demonstrated up to 20,000-*g* survival of electronics and batteries by specific fabrication/assembly techniques that are very foreign to space techniques
- Techniques for shock attenuation are generally of limited value for large, rigid-body deceleration problems (factor of 2). There is value only for spikes and possibly lateral loads. Rebound accelerations may be of significant magnitude (10%)
- State of the art in Ni-Cad batteries for high impact needs to be reestablished

CONCLUSIONS/RECOMMENDATIONS:

- Current space-electronics packaging and instrumentation mounting techniques may not withstand impacts of 1000s of *gs*
- Design studies, special analysis tools and Mars soil characteristics estimates are needed to determine design levels
- Hold a small workshop of Mars geologists and Sandia penetrator experts to develop impact models of Mars soils (i.e., S-numbers)
- Test representative electronic designs and instrumentation for impact survival plus thermal cycling, and pursue advanced technology development to correct any problems

Figure 30. Question 2

QUESTION: Are RTGs a workable power subsystem (size, location on lander)?

ASSUMPTIONS:

- Approximately 2 watts of steady state power are required at about 6 volts dc
- Thermal conductance of potential landing-site Martian soils is not known and may vary widely

TECHNOLOGY STATUS/ISSUES:

- Small 5-W PbTe thermoelectric RTGs exist for terrestrial applications. These devices operate at cold junctions of about 70°C. Heat source is not space qualified
- 250-W thermal heat source is space qualified (3 lb). The 62-thermal-watt concept (single fuel pellet) looks to be nuclear safety qualifiable if analysis can show heritage from 250-watt design
- 40-element multicouple SiGe modules have been fabricated, but impact survivability is unknown and load-direction dependent. Available voltage is only 3.5 volts
- One concept is 3 inches in diameter and 5 inches in length (2.5 lb)

CONCLUSIONS/RECOMMENDATIONS

- A 2- to 5-We RTG is probably possible within the range of 100 gs, but analysis and testing are required to prove the concept. 1000 gs survival is much less likely
- Although g loads are lower on a forebody, temperature control uncertainties make this a poor design choice given uncertainty and variability in soil conductivity
- Look at options to jettison RTG with a parachute and have it land alone at a much lower g level
- The required power/voltage levels and power subsystem design need much better definition before RTG design can proceed

Figure 31. Question 3

SUMMARY OF CONCLUSIONS AND RECOMMENDATIONS

- Consensus was that there is no clear need for new technology. Design and trade studies are required to better define technology drivers (e.g., temperature cycling, impact accelerations, RTG options). Advanced development work will most likely be required in almost every area (e.g., electronic packaging, telecomm, instrumentation, RTG) if the high-g impact penetrator concept is pursued
- Initiate immediate studies to develop temperature-control design and test representative electronic designs elements for lifetime
- There are three domains for impact environments
 - 10s of *gs* are within design experience and no advance development is needed, but a controlled (i.e., propulsive) landing is required
 - 100s of *gs* are within design experience except possibly for the RTG (or PV array) and soft landing techniques are required (e.g., large parachutes, impact attenuation, etc.)
 - 1000s of *gs* will require extensive testing and possible redesign of all electronic and instrumentation mountings and an RTG may not survive
- Depending on likely mission choices, immediately test representative electronic designs for impact survival plus thermal cycling, and pursue advanced technology development to correct any problems
- Look seriously at options such as "micro-Viking" to provide a very soft retro landing, and use alternative methods to provide soil penetration for seismometry and subsurface geochemistry/mineralogy. This might have the right kind of "novel" appeal
- Begin immediate design, analysis, and testing of RTG concepts to develop and validate a workable design
- Investigate solar-powered options, especially for "micro-Viking"
- Expect to lose some penetrators. Increase the number, and seriously consider class-D status

Figure 32. Summary of Conclusions and Recommendations

GNM Workshop Conclusions

Mission Objectives	GNM	Other
Meteorology	X	
Narrow Band Seismology	X	
Surface Chemistry	X	
Descent Imaging	X	
Landed Imaging	X (1)	
Sub-surface Chemistry	X	
Volatiles	X	
Wide Band Seismology	TBD(2)	
Aeronomy	try (3)	
Orbital Imaging		X
Other Orbital Support Observations	(4)	X
Entry Science	X	

(1) weak justification, needs work

(2) need input from seismology workshop

(3) work accommodation in FY90 spacecraft design studies

(4) a large number of participants feel this should be explored further

Figure 33. GNM Workshop Conclusions: Mission Objectives

GNM Workshop Conclusions

Payload Types	Yes	No	Number
Long life Hard Landers only		X	20+8
Long life Penetrators only		X	
Hard Landers + Short life Penetrators	X(1)		
Short life Penetrators only		X	
Add Aeronomy	?		
Other	?(2)		

Note: These conclusions are to be interpreted as FY90 priority for JPL study.

(1) also look at separate implementation of this combination, e.g., different launch vehicles, different countries, earlier launch of penetrators, use of modified CRAF penetrator for polar sites.

(2) other innovative designs may be possible, e.g., a lander with a "spike" penetrator.

Figure 34. GNM Workshop Conclusions: Payload Types

GNM Workshop Conclusions

Instruments	Surface Lander	Penetrator		Orbiter
		Forebody	Afterbody	
Aeronomy "package"				1
Meteorology station pressure temperature wind aerosols	x			2
Narrow-band seismometer	x			
Descent Imager	x	x		
Landed imager	1		1	
a-p-x	x	x	?	
DSC/EGA	x	x		
Soil Electrochemistry	x	?	x	
Neutron spectrometer	x	x		
GRS		x		
Impact accelerometer	?	x		
Entry accelerometer pressure temperature	x	x		

Other instruments TBD. This list to be updated by MarSWG workshops by April 1990 for inclusion in FY90 studies.

1 Lower priority.

2 Some participants suggest meteorology support on orbiter .

Figure 35. GNM Workshop Conclusions: Strawman Payload for FY90 Studies

GNM Workshop Conclusions

Orbit Design

Deploy on Approach (1)	<input type="checkbox"/>	yes	<input checked="" type="checkbox"/>	no
Inclination	<input checked="" type="checkbox"/>	90		
	<input type="checkbox"/>	45		
	<input type="checkbox"/>	37-39 sun-synch		
Wait for Lighting	<input checked="" type="checkbox"/>	yes	<input type="checkbox"/>	no
Aeronomy	<input type="checkbox"/>	early		
	<input type="checkbox"/>	late		
	<input checked="" type="checkbox"/>	both		

Landing Accuracy

radius, 3-sigma	desired	achievable
> 100 km	<input type="checkbox"/>	<input checked="" type="checkbox"/> hi prob
50-100 km	<input type="checkbox"/> ?	<input checked="" type="checkbox"/> maybe
10-50 km	<input checked="" type="checkbox"/> x	<input type="checkbox"/> ?
< 10 km	<input type="checkbox"/>	<input type="checkbox"/>

(1) approach deployment to be assessed for separately launched short-life penetrators.

Figure 36. GNM Workshop Conclusions: Orbit Design and Landing Accuracy

GNM Workshop Conclusions

Lander Design

Polar landers ☒ Common with Equatorial (1)
☐ Separate from Equatorial

Sub-surface ☐ None
☒ Short life Penetrator
☐ Long life Penetrator

Entry System

entry angle ☒ shallow ☐ steep
aeroshell ☒ fixed ☐ deployable (2)
orbit retro delta-V ☐ solid ☒ liquid
guided entry ☐ yes ☒ no
terminal deceleration ☒ parachute
☐ retro-rockets (3)
☒ crushable structure (4)
☒ penetrator
☐ other

- (1) may be a problem with penetration of hard ice
(2) shows promise, but riskier for 1998 launch
(3) proximity impulse solves RTG problem with high *gs*
(4) for penetrator afterbody and hard landers

Figure 37. GNM Workshop Conclusions: Lander Design

GNM Workshop Conclusions

Subsystem Technology	10-year life	high-g
Optics and Detectors		
Electronics	problem area needing advanced development	
Telecom		
Power		problem for RTGs, needs advanced development and testing
Mechanical		
Thermal and Other Environmental Control	possible problem area related to electronics and sub-surface RTGs	

Figure 38. GNM Workshop Conclusions: Subsystem Technology

SECTION 4

REFERENCES

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2. *A Robotic Exploration Program*, JPL Document D-6688, Jet Propulsion Laboratory, Pasadena, California, December 1, 1989. (JPL internal document.)
3. *Mission Concept and Development for Mars Global Network Mission*, JPL internal data package, Jet Propulsion Laboratory, Pasadena, California, December 15, 1989. (JPL internal document.)

SECTION 5

LIST OF ATTENDEES

The names, addresses, and telephone numbers of the 68 attendees are listed here. The JPL attendees are grouped together in alphabetical order and given a single address. All others are listed in alphabetical order without regard for institution or session attended.

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SECTION 6
APPENDIXES

6.1 SESSION A SUBMITTALS

Session A, Submittal No. 1

Daniel J. McCleese
Jet Propulsion Laboratory/California Institute of Technology

SCIENCE AND EXPLORATION

SCIENCE OBJECTIVES

(From a presentation by Squyres and Carr)

(Meteorology, Volatiles, Climatology, Chemistry, Mineralogy, Imaging,
Seismology, Aeronomy)

ROLE ORBITER

Aeronomy

Lower atmosphere, Climatology

Other (Imaging)



SCIENCE AND EXPLORATION

IMPLEMENTATION APPROACH

Number of Sites

Placement and Site Selection

Relationship to other missions (e.g., Sample Return)

Targeting

Landers or Penetrators

Balloons

Session A, Submittal No. 2

Francis M. Sturms
Jet Propulsion Laboratory/California Institute of Technology

Mars Global Network Mission

Objective Sets

- A. ON SURFACE, LONG LIFE
 - METEOROLOGY } Simple surface hard lander
 - NARROW BAND SEISMOLOGY }
- B. ON SURFACE, SHORT LIFE
 - SURFACE CHEMISTRY } Small delta for analysis experiments
 - DESCENT IMAGING } Add lighting constraints
 - POST-LANDING IMAGING } Small delta for survivable imaging
- C. SUB-SURFACE, SHORT LIFE
 - SUB-SURFACE CHEMISTRY } Short life penetrator
 - SUB-SURFACE VOLATILES } Delta for volatile handling
- D. SUB-SURFACE, LONG LIFE
 - WIDE BAND SEISMOLOGY } Long life penetrator
- E. ORBITAL
 - AERONOMY } Delta for S/C accommodation
 - SUPPORT IMAGING and OTHER } Delta orbit design

Mars Global Network Mission

Mission Scope Options

Option 1 - Simple Surface Hard Landers only

Single payload design, all surface objectives, medium cost

Objective sets A and B

Option 2 - Short Life Penetrators only

Single payload design, short life objectives only, minimum cost

Objective sets B and C (inadequate?)

Option 3 - Long Life Penetrators only

Single payload design, all lander objectives, most complex, max cost (?)

Objective sets A, B, C, and D

Option 4 - Simple Surface Hard Landers plus Short Life Penetrators

Dual payload designs, most lander objectives, max cost (?)

Objective sets A, B, and C

Any Above Option plus one or more Orbital Objectives

Adds S/C complexity and cost

Adds Objectives from Set E

Mars Global Network Mission

Fran's Proposed Mission for Discussion

- Long Life Surface Hard Landers only (Option 1) plus Aeronomy
(meets many objectives, limits cost)
- Delay Sub-surface Objectives for Rover Missions
(rovers can get to volatiles, better platforms for drilling and analysis)
- Perform Support Imaging with Mars Observer Backup
(a good reason to do this mission)
- Deploy approximately 20 landers from polar, 1/5 sol orbit to all latitudes
(no approach deployments)
- Lower periapsis before and/or after lander deployment for aeronomy
(about 6 months to first lander deploy)

Session A, Submittal No. 3

David Morrison
Ames Research Center

MARS GLOBAL NETWORK MISSION SCIENCE REQUIREMENTS OVERVIEW

**Comments by David Morrison
NASA Ames Research Center
6 February 1990**

MISSION MUST MEET BOTH EXPLORATION AND SCIENCE OBJECTIVES

Exploration requirements set by HEI (OAST/OEXP)

Science requirements defined by Mars SWG

Mission design and execution carried out by OSSA

NASA

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Space Science

Division

MARS GLOBAL NETWORK MISSION SCIENCE REQUIREMENTS OVERVIEW

Morrison page 2

MISSION SHOULD BE FOCUSED ON NETWORK-UNIQUE OBJECTIVES

The Network Mission is one of a sequence of robotic missions to Mars
Objectives of first three robotic missions are global characterization of Mars
Other missions will provide capability for remote sensing from orbit
Other missions will provide surface sample return
Other missions will provide surface rovers with analysis capabilities

Conclusion: Limit the objectives of this mission to those things for which network (many distributed sites) is enabling factor

Also: Anticipate that the network objectives will be supported by other missions; i.e., supplement these surface stations with others deployed by other missions



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MARS GLOBAL NETWORK MISSION SCIENCE REQUIREMENTS OVERVIEW

Morrison page 3

FOUR CLASSES OF SCIENCE AND EXPLORATION OBJECTIVES

- 1. Interior structure: Seismometry**
- 2. Atmospheric circulation: Meteorology**
- 3. Soil properties and exobiology: Physical and Chemical Analysis**
- 4. Surface imaging: Approach imaging and in situ imaging**

INTERPRETATION

- No. 1 & 2 require long lifetime; 3 & 4 do not**
- No. 3 requires subsurface access**
- No. 2, 3, and 4 relate directly to HEI objectives**
- No. 3 will require most careful pruning of possible experiments**
- No. 4 may be satisfied by other missions (i.e., Mars 94 balloon)**



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Session A, Submittal No. 4

Michael H. Carr
U.S. Geological Survey

	METEOROLOGY	VOLATILES	CHEM/MIN	IMAGING	SEISMIC HIGH FREQUENCY	SEISMIC LOW FREQUENCY	GEOTECH
BELOW GROUND		✓	✓		OK	✓	✓
ON GROUND	✓	OK	✓	✓	OK		
NUMBER OF STATIONS	~20	6	~10	10+	10-20	3	
LIFETIME	>1 yr	W	W	W	>>1 yr	>>1 yr	Sec
ELLIPSE ERROR	600 km	60 km	60 km	60 km	60 km Sep ^w	>1000 km Sep ^w	
SAMPLING CAPABILITY	—	✓	✓				
POLAR	✓	✓					✓
DATA VOLUME	LO	LO	LO	HI	HI	HI	LO

SCIENCE OBJECTIVES

- WHAT ARE VALID SCIENCE OBJECTIVES FOR THIS MISSION?
- OBJECTIVES INVOLVING:
 - MEASUREMENTS IN A WIDE VARIETY OF GEOLOGIC TERRAINS
 - MEASUREMENTS IN LOCATIONS THAT CANNOT BE REACHED BY OTHER MEANS (E.G., TOO HIGH LATITUDE, TOO RUGGED, ETC.)
 - SIMULTANEOUS GLOBAL OBSERVATIONS
- GLOBAL SEISMIC NETWORK
 - DEEP INTERNAL STRUCTURE
 - CRUSTAL STRUCTURE
 - SEISMIC ACTIVITY
- GLOBAL METEOROLOGICAL NETWORK
 - GLOBAL CIRCULATION
 - DUST AND WATER TRANSPORT
 - BOUNDARY LAYER

SCIENCE OBJECTIVES_(CONT'D)

- VOLATILES
 - GLOBAL INVENTORY/DISTRIBUTION
 - POLAR DEPOSITS
 - RESOURCES
- GEOCHEMISTRY/MINERALOGY
 - ROCK TYPES/PHASE ASSEMBLAGES/PETROGENESIS
 - GROUND TRUTH FOR REMOTE SENSING
 - EMPHASIS ON BEDROCK
- SURFACE MORPHOLOGY
 - GEOLOGIC STUDIES
 - HAZARD ASSESSMENT
 - CONTEXT FOR OTHER MEASUREMENTS
- REGOLITH STRUCTURE

Session A, Submittal No. 5

Bruce Murray
California Institute of Technology

James D. Burke
Jet Propulsion Laboratory/California Institute of Technology

BALLOONS FOR ENHANCED SCIENTIFIC RETURN FROM MARS HARD LANDERS

by

Bruce Murray, Caltech, and James D. Burke, JPL

Assisted by:

Bruce Betts, James Consolver, Laszlo Keszthelyi,
and Tom Svitek, Caltech; Robert Mostert, JPL;
Tom Heinsheimer, Titan Systems

February 6, 1990

I. SCIENTIFIC OBJECTIVES

IMAGING

- Same scientific objectives as descent imaging, i.e., sample of small scale features plus an engineering description of terrain.
- Low-altitude balloons offer an alternative to descent imaging, eliminating local time of day and probe - orbiter constraints on mission.
- Higher resolution, greater coverage and multiple lighting coverage can be achieved with low-altitude balloons.

BOUNDARY LAYER SAMPLING

- Sampling needed: (1) to tie Mars GCM to surface; (2) to ascertain magnitude of future descent/ascent challenge; (3) to validate models of dust/H₂O concentrations; and (4) to perhaps obtain H₂O vertical flux.
- Balloons can provide: (1) direction and speed of 100 meter altitude winds; (2) temperature at 100 meters versus surface; (3) possibly humidity at 100 meters versus surface, possibly H₂O flux; and (4) possibly sample dust content at 100 meters.

II. TECHNICAL BASIS

VEGA BALLOON

- Development background.

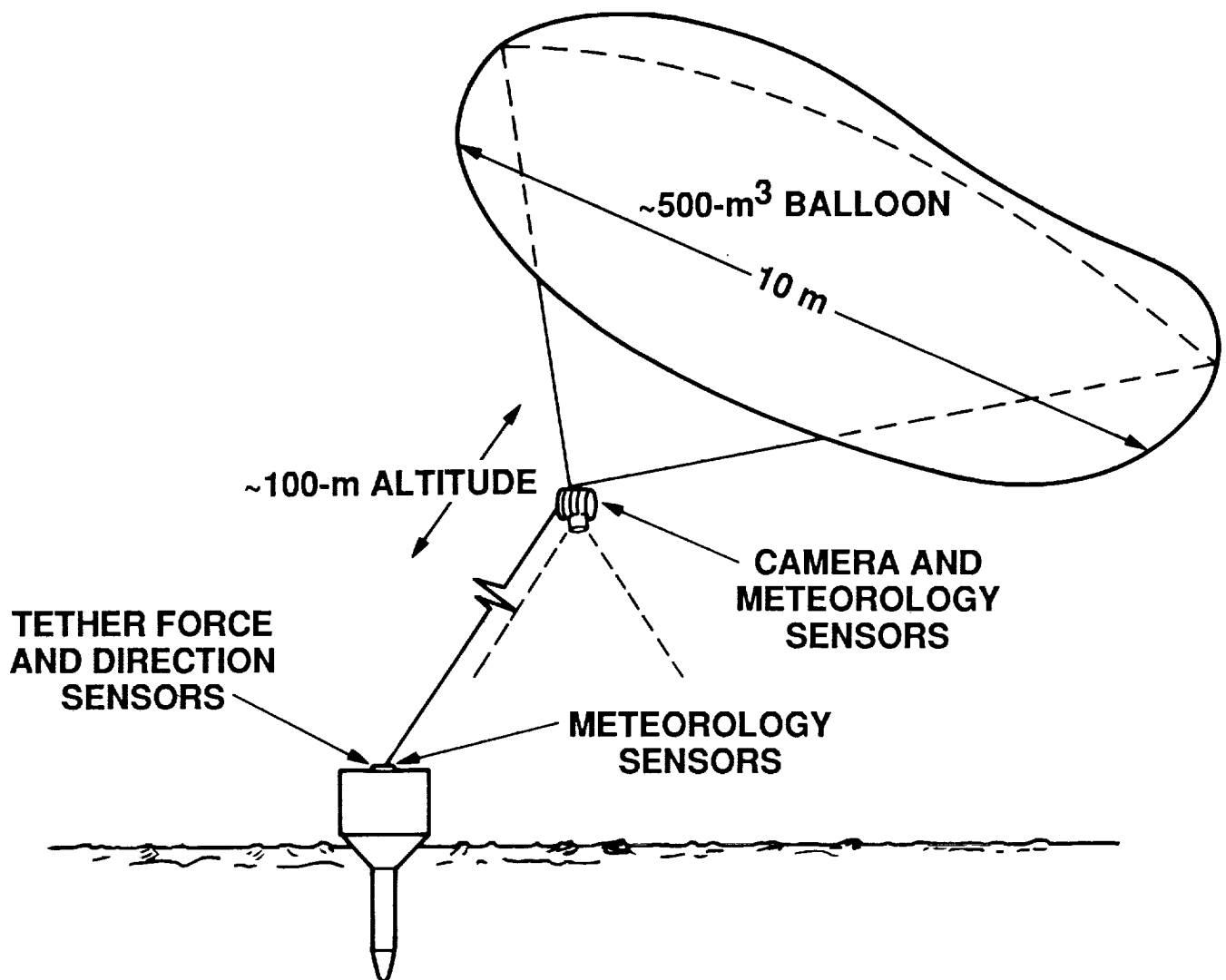
MARS '94

- 10 times larger balloon planned to be deployed by USSR/CNES.
- Snake development under way.

POST '94

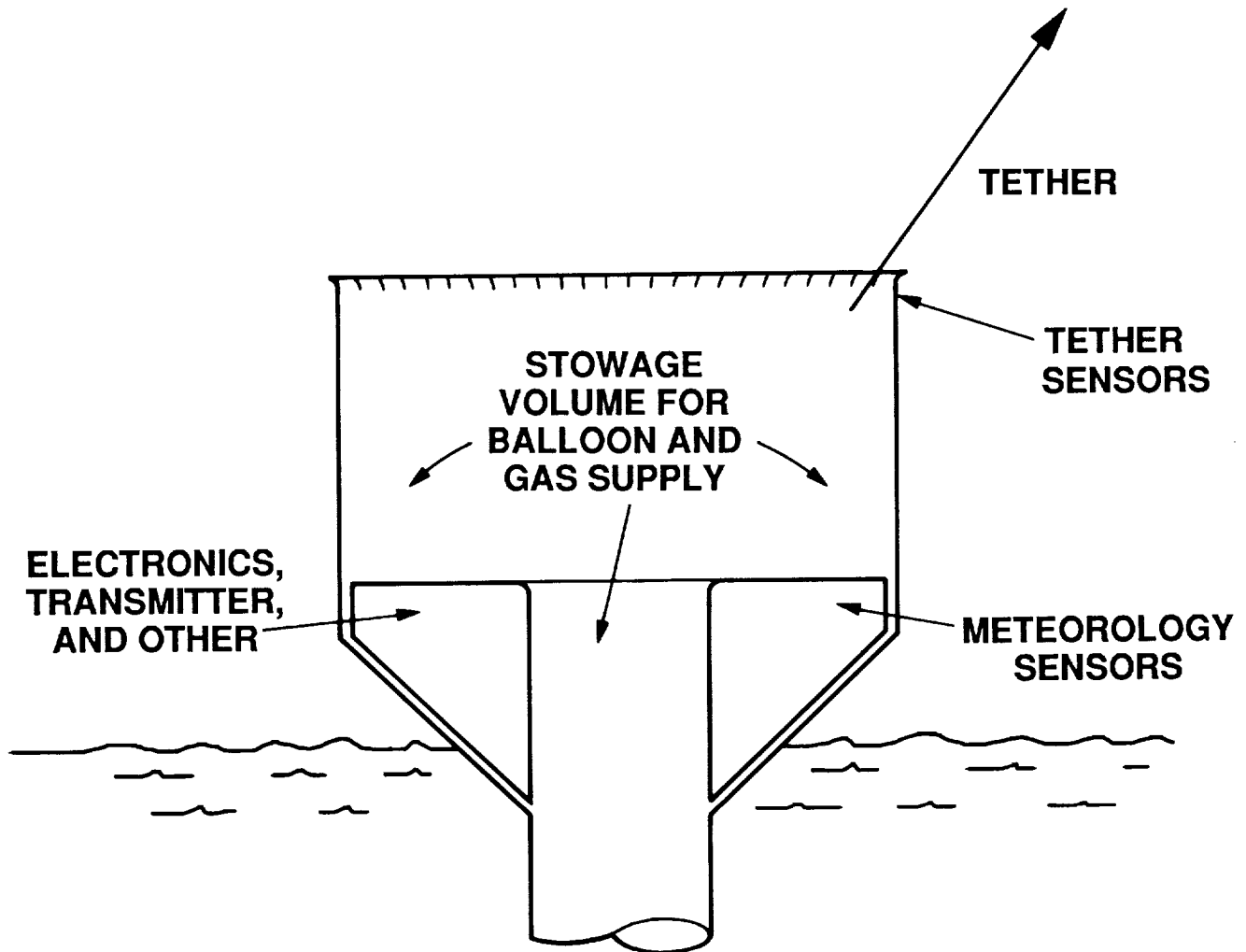
- Descent imaging camera technology applicable to balloon.

JPL TETHERED BALLOON CONCEPT



JPL

BALLOON ANCHORING LANDER CONCEPT



SAMPLE AIRBORNE PAYLOAD

CAMERA

- Optics	80 gms
- Electronics	140 gms
- Sun sensors, etc.	80 gms
	300 gms

METEOROLOGICAL PACKAGE

- temperature, slip speed, dust/H ₂ O detector	50 gms
--	--------

GONDOLA

- Structure	100 gms
- Radio relay	50 gms
- Batteries (50 watt-hrs)	200 gms
- Thermal (RHU, insulation)	50 gms
	400 gms

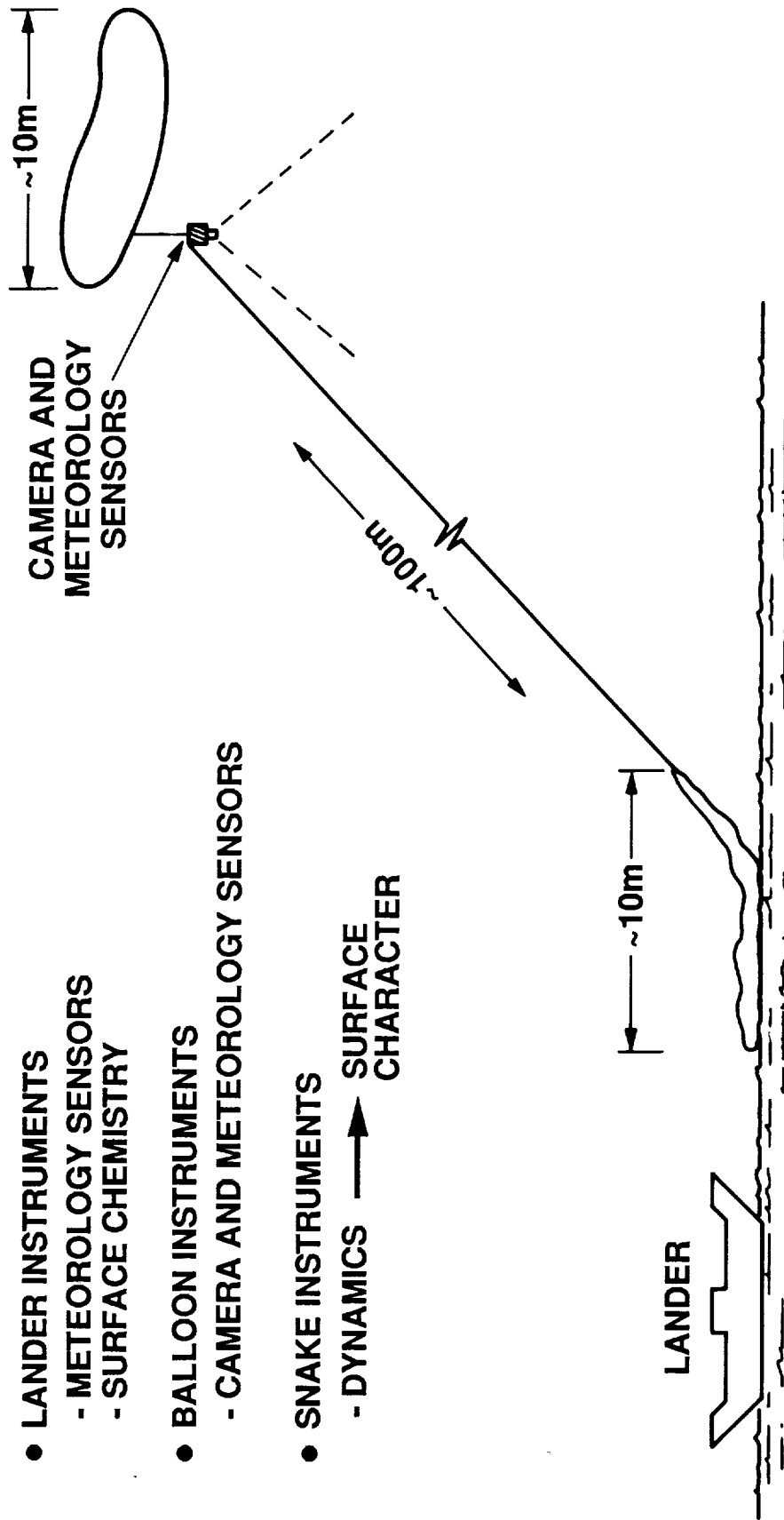
CONTINGENCY (33%)

250 gms

TOTAL

1000 gms

JPL CONCEPT FOR HARD LANDER WITH BALLOON AND SNAKE



CONCLUSIONS

- Small low-altitude balloons may significantly enhance scientific value of landers.
- Low-altitude balloon imaging better than descent imaging and balloon gives only way to do boundary-layer measurements.
- However, tethering to fixed lander gives novel problems.
 - post-landing deployment
 - relative wind speed at balloon
 - tether pull on lander
- Snake concept relieves balloon and lander problems, also provides measurement profiling.
- We recommend inclusion of balloon/snake as a payload option.

Session A, Submittal No. 6

David Morrison
Ames Research Center

MARS GLOBAL NETWORK MISSION STRAWMAN LANDER PAYLOAD

- **METEOROLOGY**
 - **PRESSURE**
 - **DUST LOAD (DIRECT AND SCATTERED LIGHT)**
- **SEISMOLOGY**
 - **HIGH-FREQUENCY SEISMOMETER**
- **SURFACE CHEMISTRY**
 - **$\alpha/p/x$ (MAJOR ELEMENT CHEMISTRY)**
- **SOIL PHYSICS**
 - **ACCELEROMETER**
- **IMAGING**
 - **to be determined**

Session A, Submittal No. 7

Steven W. Squyers
Cornell University

PENETRATORS OR HARD LANDERS?

PENETRATORS

- **PROS:**
 - **ENABLE DEEP SAMPLING**
 - **SOME HERITAGE (MILITARY, CRAF)**
- **CONS:**
 - **SEVERE CROSS-SECTION CONSTRAINTS**
 - **RTG HEAT MAY POSE A PROBLEM**
 - **HIGH *gs* REQUIRED FOR PENETRATION**

HARD LANDERS

- **PROS:**
 - **SIMPLER INSTRUMENT ACCOMMODATION**
 - **SIMPLER HEAT REJECTION**
- **CONS:**
 - **DEEP SAMPLING *VERY* DIFFICULT**
 - **LITTLE HERITAGE**

**APPARENT NATURAL DIVISION INTO
SHORT-LIVED/SUBSURFACE VEHICLES AND
LONG-LIVED SURFACE VEHICLES**

TWO VEHICLE TYPES

A) ~8 SHORT-LIVED PENETRATORS

- **GRS (NaI; Ge POLAR?)**
- **α -p-x**
- **n-SPECTROMETER/14 MeV SOURCE**
- **DSC/EGA**
- **ACCELEROMETER**
- **DESCENT IMAGING**

B) ~16 LONG-LIVED HARD LANDERS

- **SEISMOMETER**
- **METEOROLOGY PACKAGE**
- **DESCENT IMAGING, α -p-x**

***BOTH REQUIRE ACCURATE TARGETING AND
MINIMAL LATITUDE/LONGITUDE RESTRICTIONS***

THREE KEY QUESTIONS

- (1) TO WHAT EXTENT DOES *SURFACE* EMPLACEMENT DEGRADE SEISMIC SCIENCE?
- (2) TO WHAT EXTENT DOES SURFACE EMPLACEMENT DEGRADE GEOCHEMISTRY/VOLATILES SCIENCE?
- (3) HOW MANY STATIONS ARE *REALLY* REQUIRED FOR METEOROLOGY AND SEISMOLOGY?

Session A, Submittal No. 8

William B. Banerdt
Jet Propulsion Laboratory/California Institute of Technology

PAYLOAD FOR A MARS GLOBAL NETWORK MISSION

Ground Rules for "Pre-Strawman" Selection

ORBITER

Payload Capacity:	Mass	?
	Power	?
	Telemetry Rate	?
Orbit:	Circular or Elliptical?	?
	High or Low Periapsis?	?

LANDER

Payload Capacity:	Mass	?
	Power (Peak, Sustained)	?
	Telemetry Rate (Data Storage)	?
Type of Lander:	Hard or Soft Lander, Penetrator?	?
Number:	Few (~6) or Many (~20)?	?
Lifetime:	Short (days) or Long (years)?	?

PAYLOAD FOR A MARS GLOBAL NETWORK MISSION

Surface Science

Long Lifetime (2–10 years)

Wide-Band Seismometer

Meteorology: **Pressure**
Temperature
Wind (Speed and Direction)
Aerosols (Solar Extinction)
Humidity

Radio Beacon (Tracking)

Magnetometer (Aeronomy)

Short Lifetime (Hours to Days)

Soil Characterization: **Gamma-Ray Spectrometer**
Alpha, Proton, X-Ray Spectrometer
Differential Scanning Calorimeter
Evolved Gas Analyzer/Gas Chromatograph
Neutron Spectrometer
Soil Chemistry Analyzer (eH,pH)

Impact Accelerometer

Panoramic Camera

PAYLOAD FOR A MARS GLOBAL NETWORK MISSION

Orbiter and Entry Science

ORBITER SCIENCE

Radio Science:

**X Band (S Band) Transponder
Stable Oscillator**

Aeronomy Package:

**Neutral Mass Spectrometer
Ion Mass Spectrometer
Retarding Potential Analyzer/Ion Driftmeter
Langmuir Probe
Plasma/Energetic Particle Analyzer
Plasma Wave Analyzer
Magnetometer
Fabry-Perot Interferometer
UV/IR Spectrometer**

Meteorology Package:

**Synoptic Camera (Visual/UV)
IR Atmospheric Sounder**

ENTRY SCIENCE

Nested Approach Imager

Pressure/Temperature

Accelerometer

Mass Spectrometer

PAYLOAD FOR A MARS GLOBAL NETWORK MISSION

Instrument	Mass (kg)	Power (W)	Data Rate
<u>Lander</u>			
Accelerometer (Impact)	0.15	1.	2 kb
Accelerometer (Descent)/Altimeter	2.	4.	?
Aerosols	0.1	0.1	20 kb/day
Alpha, Proton, X-Ray Spectrometer	0.25	0.15	12 kb/sample
Camera (Descent)	0.4	1.	2 Mb/image
Camera (Panoramic)	0.5	0.5	20 Mb/image
Differential Scanning Calorimeter	1.2	15.	2 Mb/sample
Evolved Gas Analyzer/Gas Chromatograph	3.6	7.	2 Mb/sample
Gamma-Ray Spectrometer	1.3	0.8	12 bps
Humidity	0.3	0.25	50 b/hr
Magnetometer	0.5	0.2	2 kb/hr
Mass Spectrometer	10.	15.	?
Neutron Spectrometer	?	?	?
Pressure/Temperature/Wind (Surface)	0.3	0.75	10 bps
Pressure/Temperature (Descent)	0.2	0.5	500 kb
Radio Beacon	?	?	—
Seismometer	0.5	0.3	10 Mb/day
Soil Chemistry Analyzer (Oxidation)	0.25?	0.2?	12 kb?
<u>Orbiter</u>			
Fabry-Perot Interferometer	13.5	5.5	30 bps
Ion Mass Spectrometer	2.5	1.5	60 bps
IR Atmospheric Sounder (MAO)	8.	7.5	260 bps
(PMIRR)	25.	25.	150 bps
Langmuir Probe	2.	4.	30 bps
Magnetometer	3.	3.5	200 bps
Neutral Mass Spectrometer	10.	8.5	180 bps
Plasma Wave Analyzer	5.	3.5	130 bps
Plasma/Energetic Particle Analyzer	10.	9.	320 bps
Retarding Potential Analyzer/Ion Driftmeter	4.5	4.	80 bps
Stable Oscillator	1.6	2.8	—
Synoptic Camera (Visual/UV)	9.	8.	1000 bps
UV/IR Spectrometer	5.	7.	130 bps

PAYLOAD FOR A MARS GLOBAL NETWORK MISSION

	<u>Mass (kg)</u>	<u>Power (W)</u>	<u>Telemetry (bps)</u>	<u>Data Storage (Mb)</u>
ORBITER	56.1	52.3	2260	—
LANDER (Peak)	3.45	2.5	—	20 (10 images)
LANDER (Cont.)	3.45	2.3	140	12 (1 day)

Session A, Submittal No. 9

Janet Luhmann
University of California at Los Angeles

Vu-graph Captions

- 1) A list of science objectives which point to the value of a Mars aeronomy mission to an overall program of understanding Mars and its environment.
- 2) Altitude profiles of the neutral upper atmosphere and ionosphere obtained with the Viking Landers.
- 3) Example of radio-occultation derived electron density profiles of the Martian ionosphere from Mariner 9.
- 4) Altitudes of the peak electron densities versus solar zenith angle showing the effects of dust storm activity. Dust storms heat the lower atmosphere and thereby raise the density in the upper atmosphere - causing the ionosphere to form at higher altitudes.
- 5) Illustration of the small size of the Mars-solar wind interaction region compared to that of the Earth. Earth's relatively strong magnetic field creates a large magnetic bubble which protects the atmosphere and ionosphere from direct interaction with the solar wind.
- 6) Comparison of a model of the Martian ionosphere (Shinagawa and Cravens, 1989) with data from Viking (Hanson et al., 1977). Notice that the top of the ionosphere appears to be "removed" in the observed profile. This is one effect of solar wind scavenging.
- 7) One mechanism by which the solar wind can remove ions is through MHD (magnetohydrodynamic) forces associated with the interplanetary magnetic field, which "hangs up" on the conducting obstacle of the ionosphere.
- 8) Mars also has an extended neutral upper atmosphere (exosphere) of atomic oxygen produced by the photochemistry at lower altitudes. Hydrogen is also present at high altitude.
- 9) The solar wind "picks up" ions created from those regions of the neutral exosphere that extend out into the flowing solar wind and magnetosheath plasmas. Some are removed, while others reimpact the dayside.
- 10) Some flux levels of O^+ ions at various energies expected in the vicinity of Mars from the pick-up process (from a model by Luhmann, 1990).
- 11) In addition to ion pick-up, planetary atmosphere particles escape by virtue of at least two other processes. Some of the neutral atoms simply have upward-directed velocities greater than the ~ 5 km/s escape velocity. Other neutrals escape because they are "sputtered" from the atmosphere by the pick-up ions (Luhmann and Kozyra, 1990).
- 12) Energy spectrum of picked-up O^+ ions precipitating into the dayside atmosphere of Mars from a model by Luhmann and Kozyra (1990).
- 13) Upgoing neutral oxygen atom spectrum from the normal nonthermal escape mechanism (top) and with the sputtered population (bottom).

- 14) Illustration of magnetic fields in the dayside ionosphere of Venus as seen by the Pioneer Venus Orbiter (left) and as modeled (right) by a diffusion/convection calculation. The Martian ionosphere may everywhere have fields similar to those in the subsolar region of Venus (region I).
- 15) When the subsolar region of the Venus ionosphere is magnetized, it is "disturbed," with electron density fluctuations of magnitude shown here in panel (V) on the left (from Woo et al., 1988).
- 16) The geometry of these observations at Venus is as illustrated here. The spacecraft was submerged in the ionosphere and transmitting to Earth when these disturbances were detected.
- 17) Further detail on the appearance of the ionospheric disturbances in the Doppler shift of the telemetry signal. The bottom panels have been corrected for the expected Doppler shifts. These kinds of disturbances may also occur in transmissions through the subsolar region of the Martian ionosphere.
- 18) NASA documentation in support of a Mars Aeronomy Mission (MAO) to address these and other science objectives exists in the form of a report prepared by JPL. This is the cover sheet.
- 19) The strawman payload recommended in the MAO report.
- 20) A description of the strawman payload instruments.
- 21) Cross-correlation of science objectives and proposed MAO instruments.
- 22) The Mars Network Mission Orbiters may provide the vehicle for carrying out an effective MAO mission.
- 23) The availability of two spacecraft is of tremendous benefit if one can monitor the solar wind while the other makes measurements of the Martian system.
- 24) Some desirable characteristics of MAO spacecraft are included in this list.
- 25) The periapsis altitude would determine what science could be done.
- 26) The current Mars Network Spacecraft design would need to somehow incorporate the features of an aeronomy spacecraft like the Pioneer Venus Orbiter on the probe carrier.
- 27) Further description of the PVO spacecraft, showing the desirable characteristics of unobstructed ram-face instruments, a magnetometer boom, and body-mounted solar cells.

Mars Aeronomy Mission Science Objectives

Upper Atmosphere:

- **properties and variability (e.g., response to dust storms, seasons, solar activity, solar wind conditions)**
- **loss processes/evolution**

Ionosphere:

- **Source of Nightside Ionosphere (e.g., auroral activity?)**
- **temporal and spatial variability/disturbances (e.g., response to dust storms, seasons, solar activity, solar wind conditions)**

Magnetic Field:

- **nature/origin**
- **variability**
- **effects on energetic particle (radiation) environment**

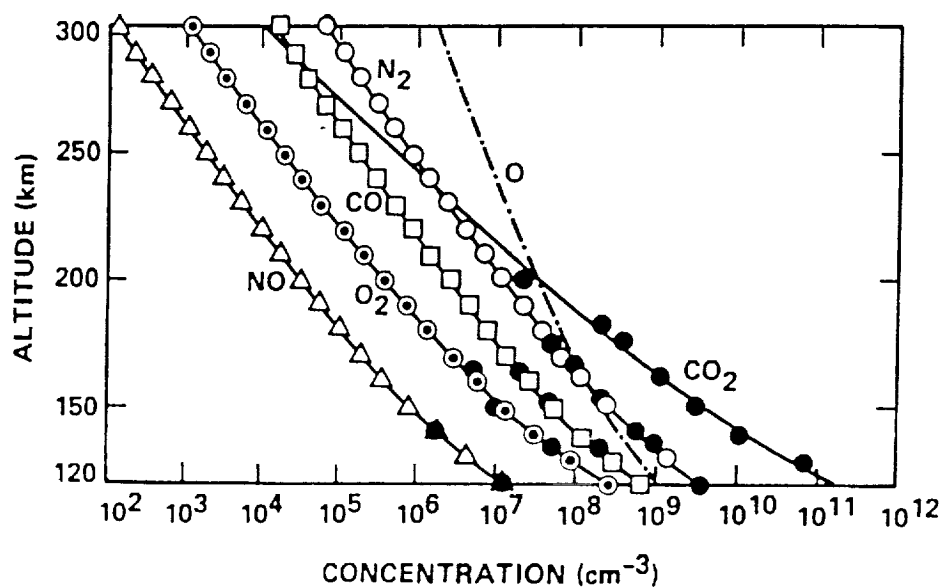
Solar Wind Interaction:

- **significance of planetary magnetic fields**
- **comparisons with Venus and Earth**

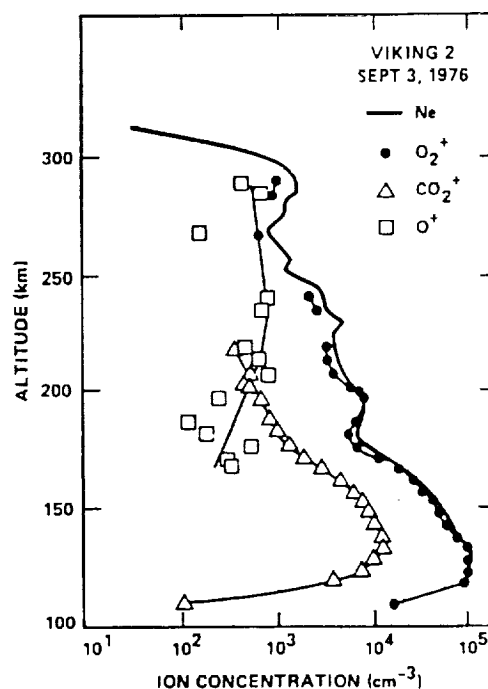
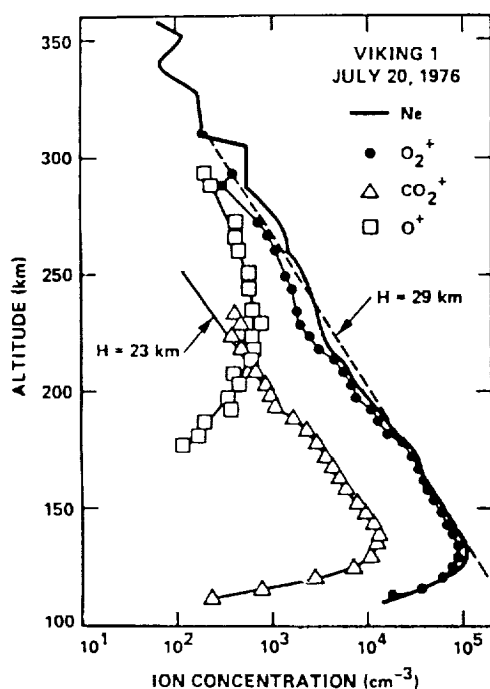
Vu-graph 1

AERONOMY RESULTS FROM VIKING LANDERS

(neutral atmosphere)

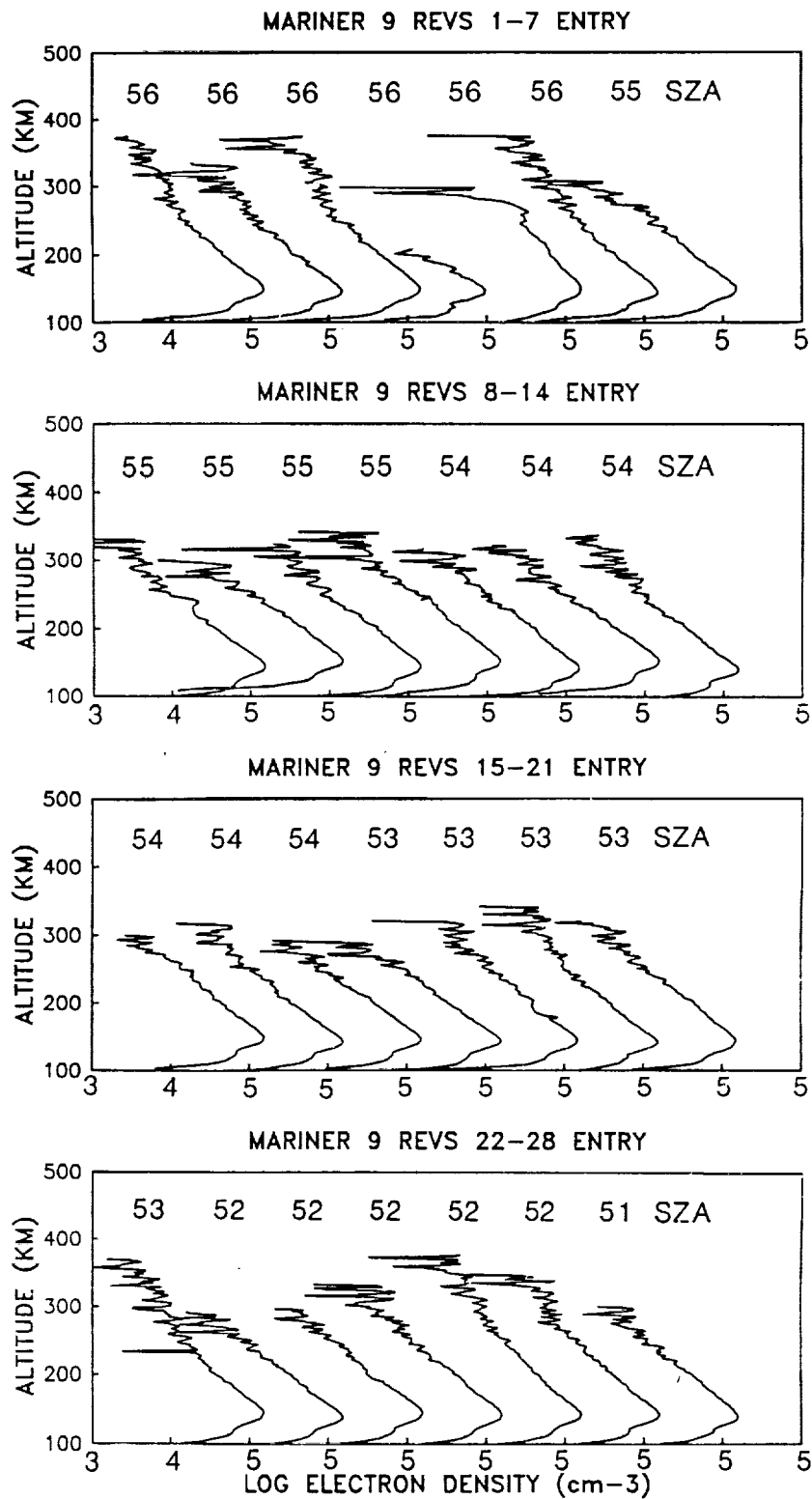


(ionosphere)



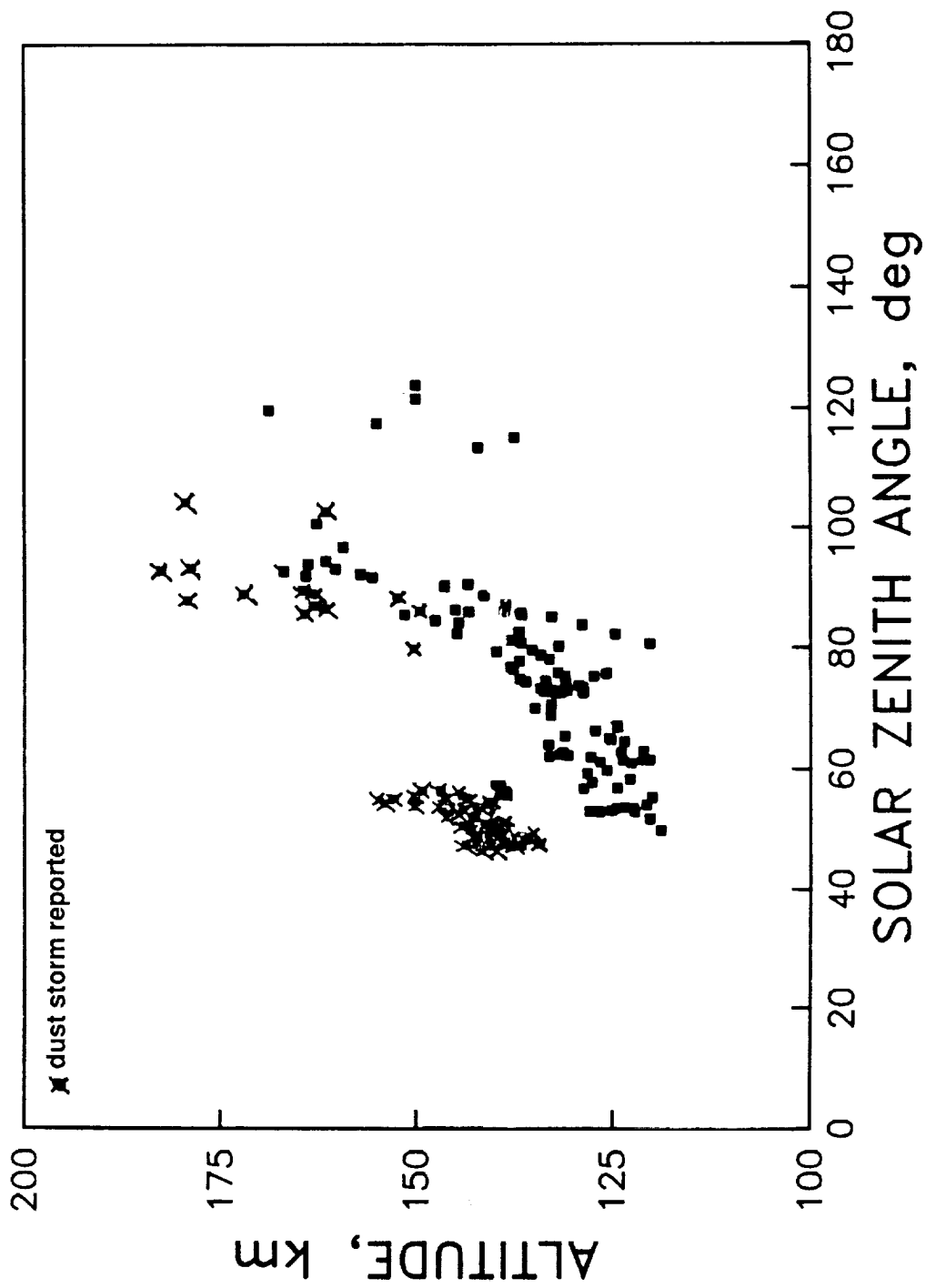
Vu-graph 2

RESULTS FROM RADIO OCCULTATION EXPERIMENTS

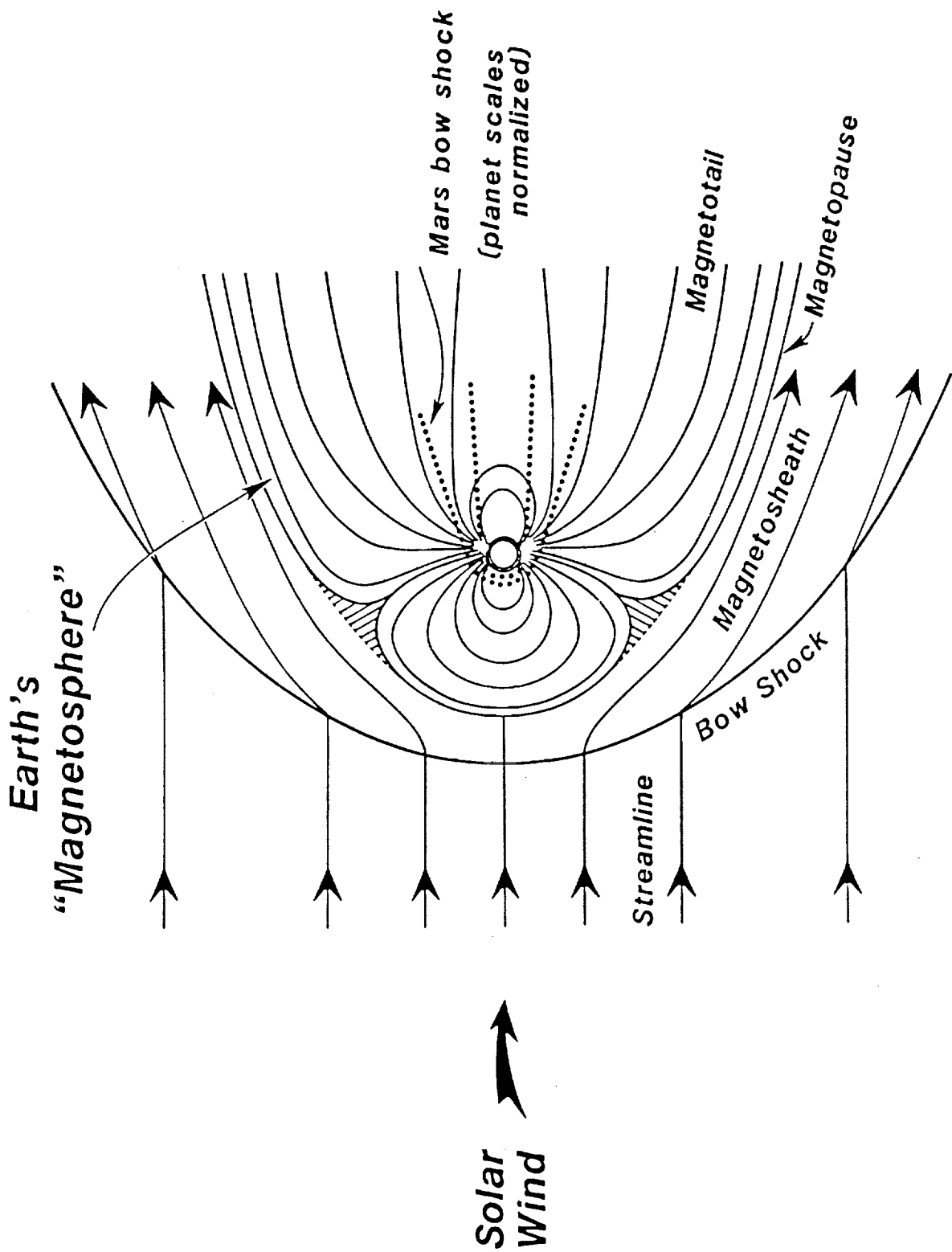


Vu-graph 3

Mars

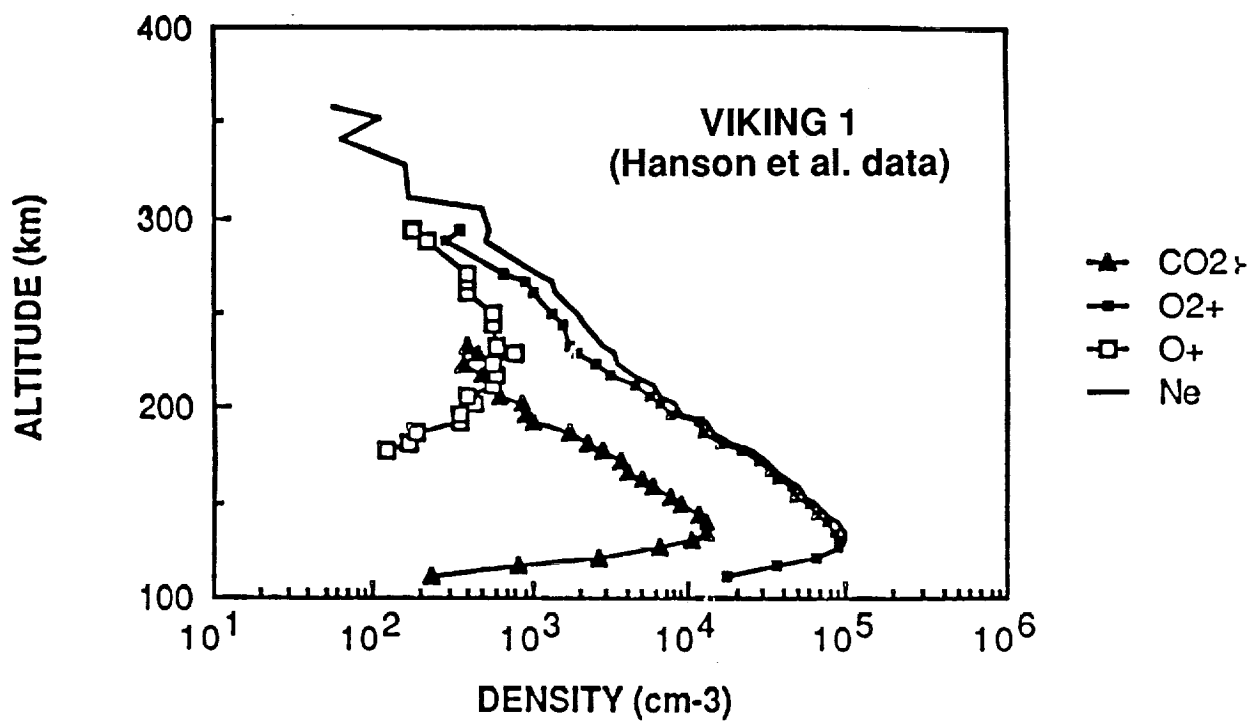
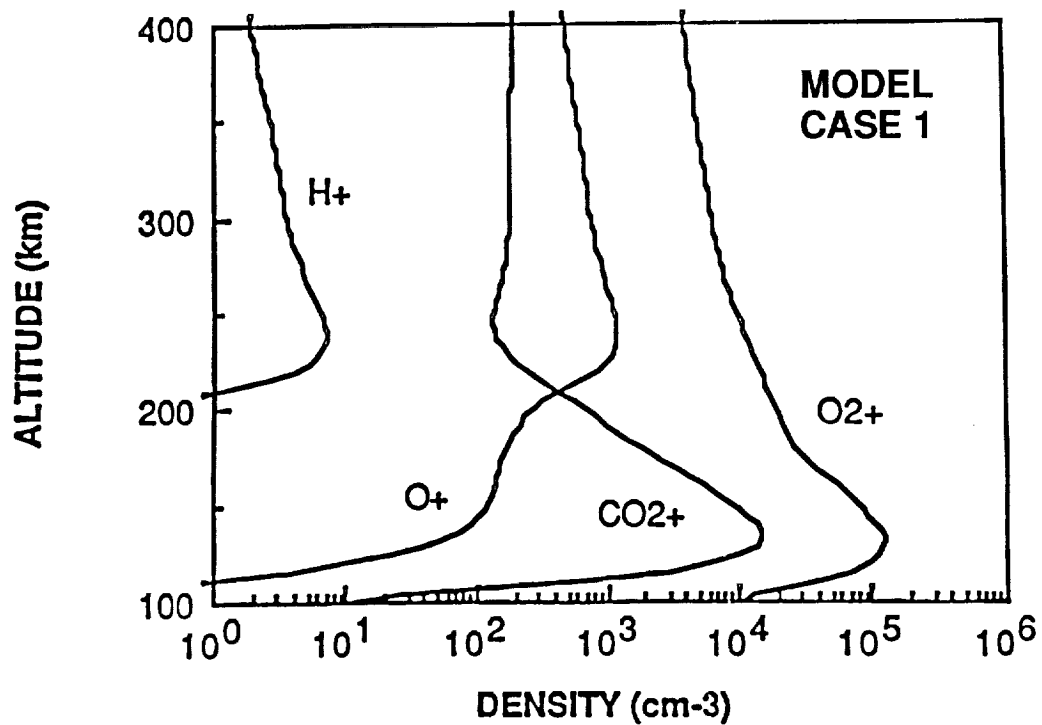


Vu-graph 4



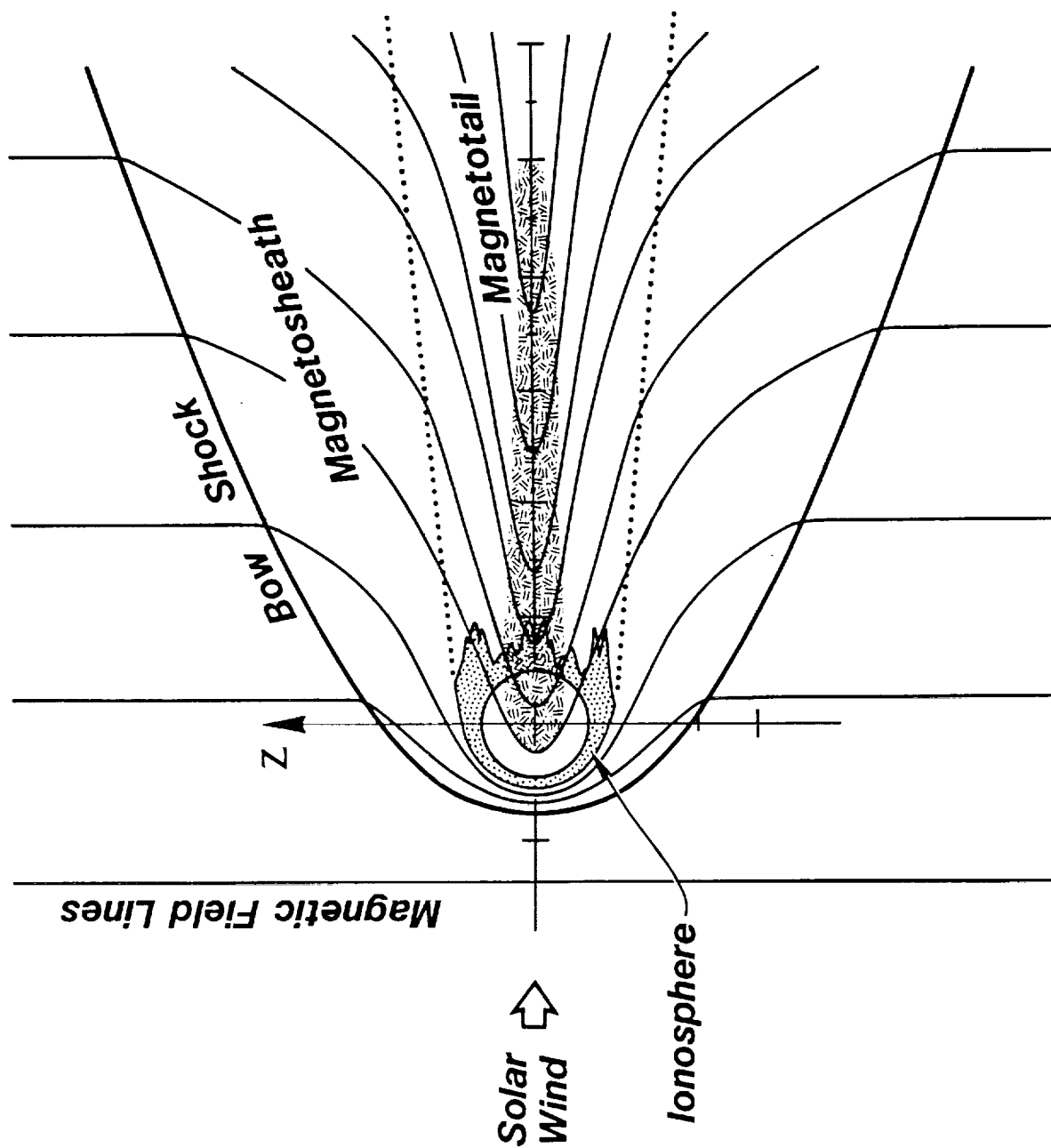
Vu-graph 5

MARS IONOSPHERE

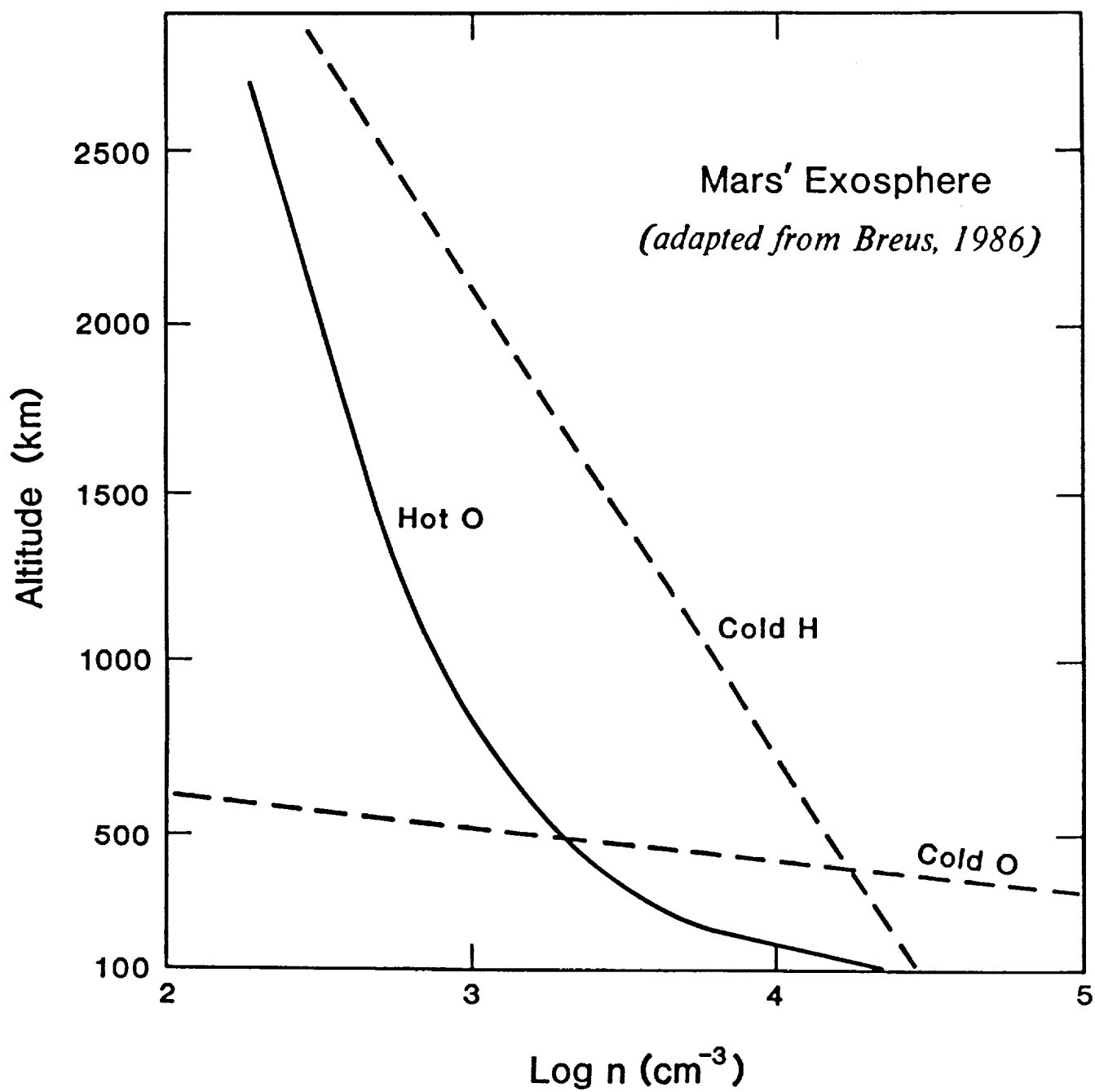


From Shinagawa (1989)

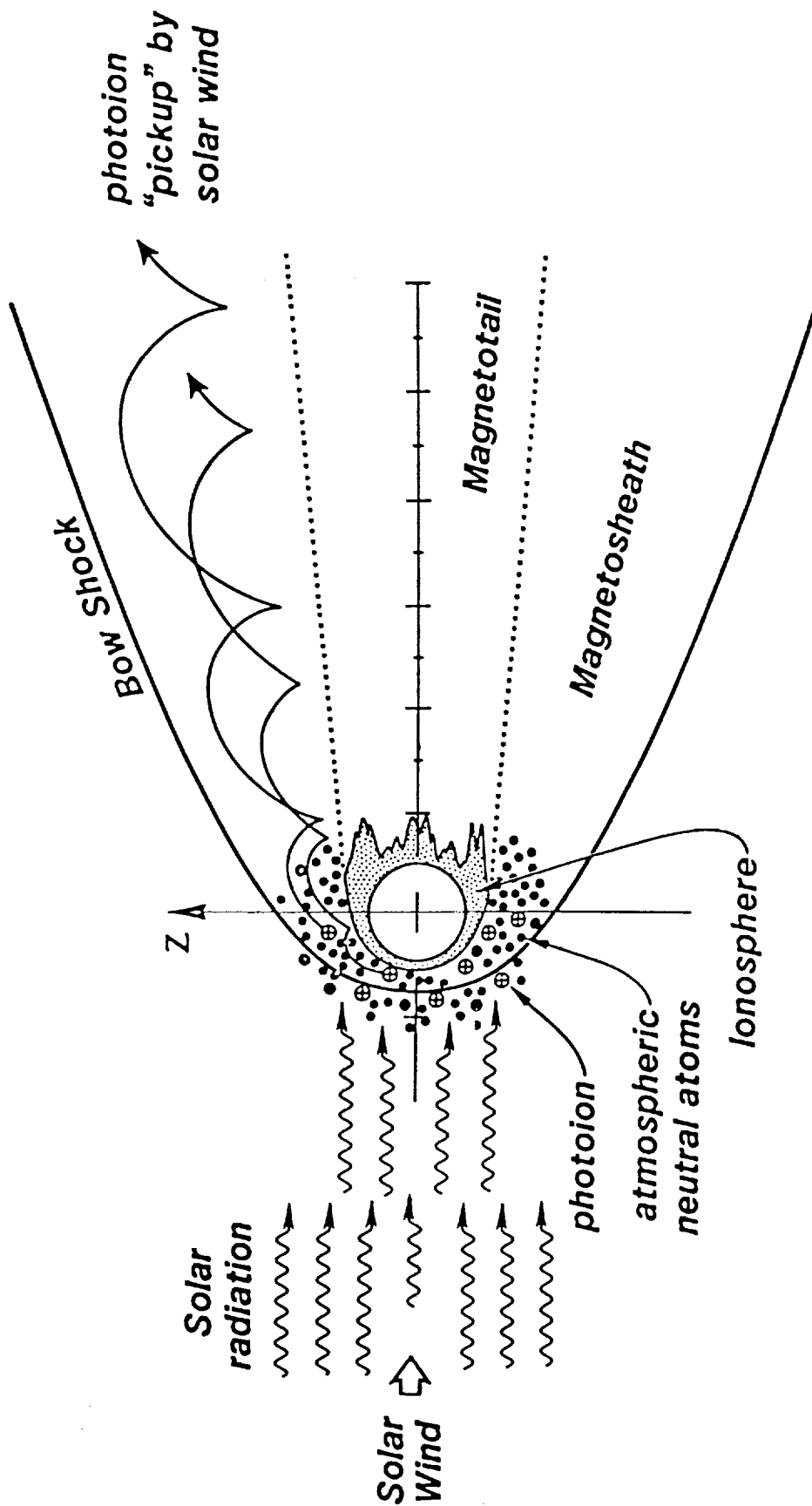
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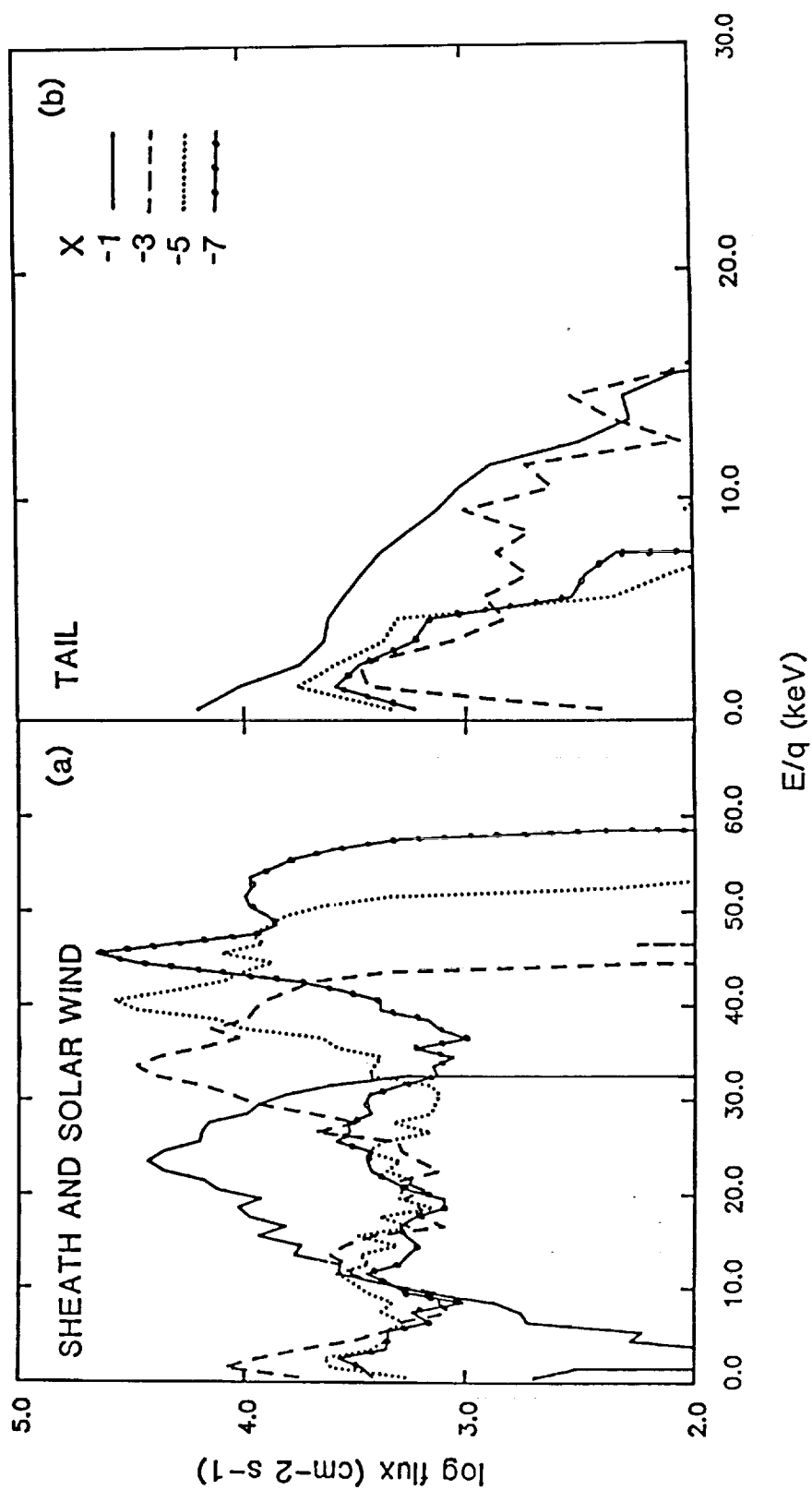
Vu-graph 7



Vu-graph 8

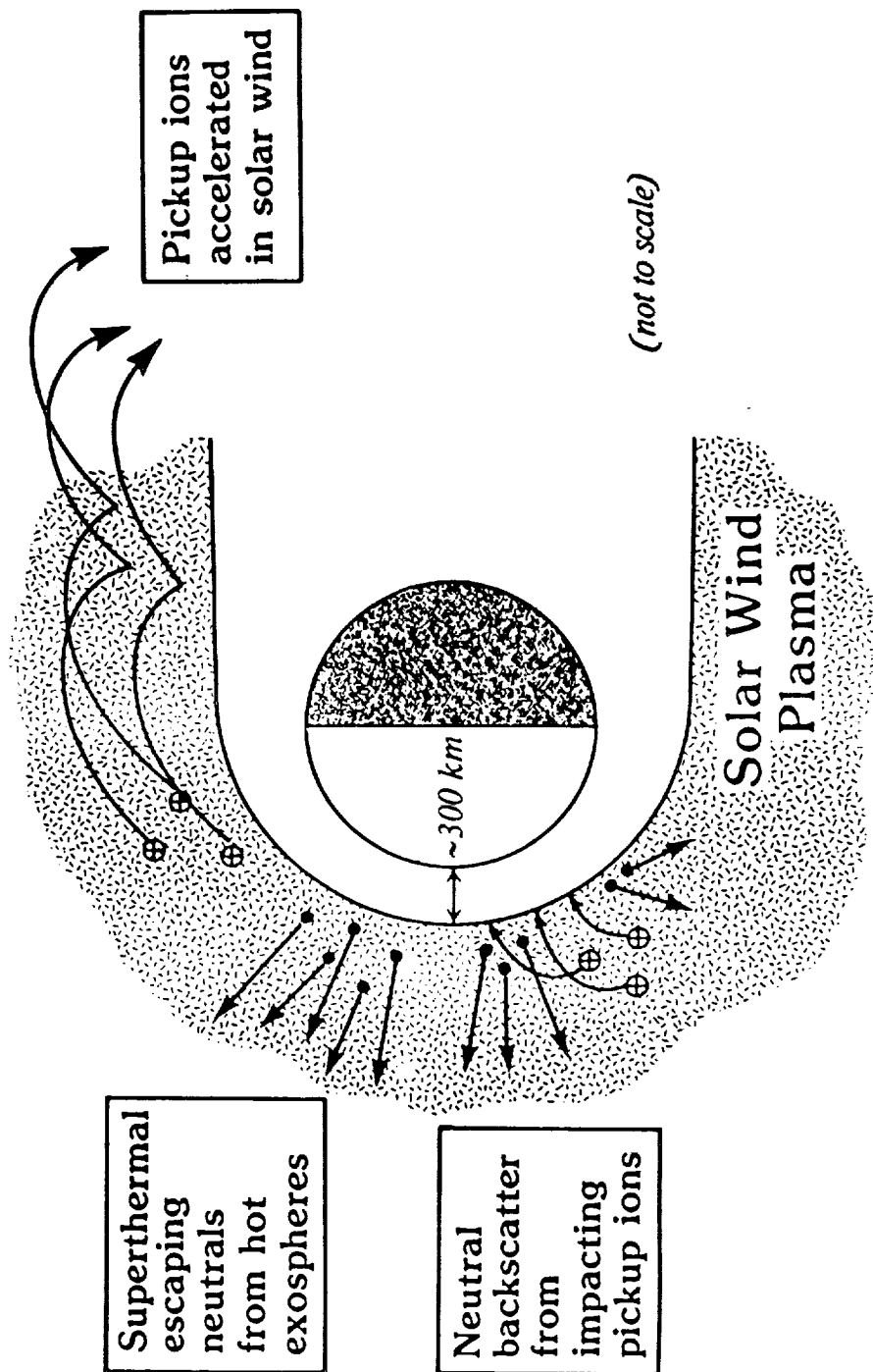


Vu-graph 9

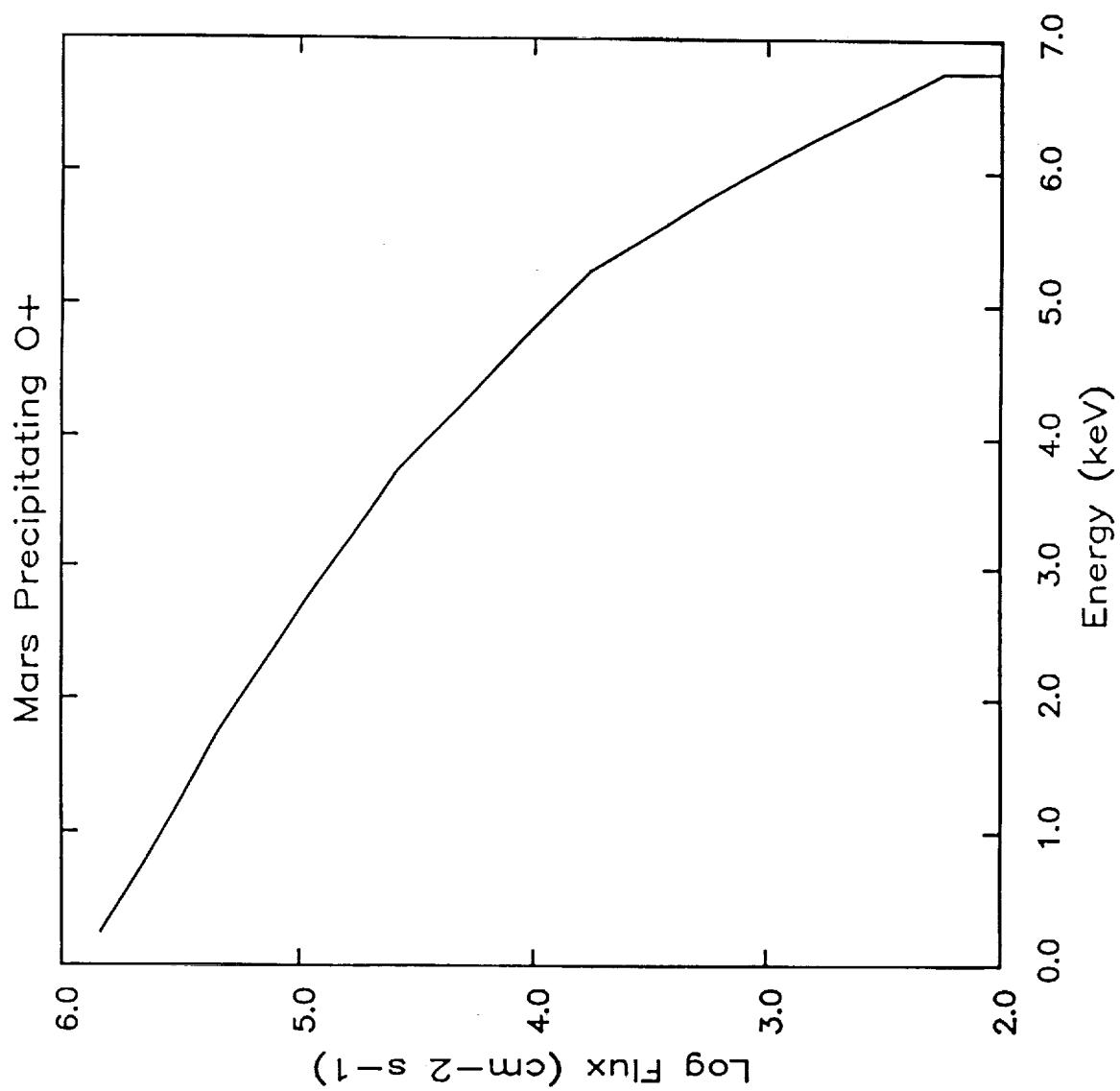


Vu-graph 10

Nonthermal Escape - Venus and Mars

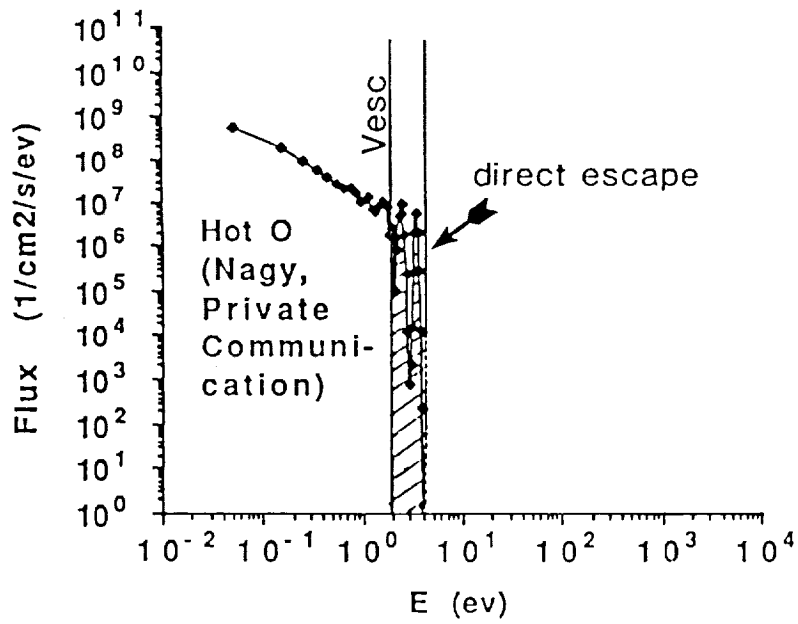


Vu-graph 11

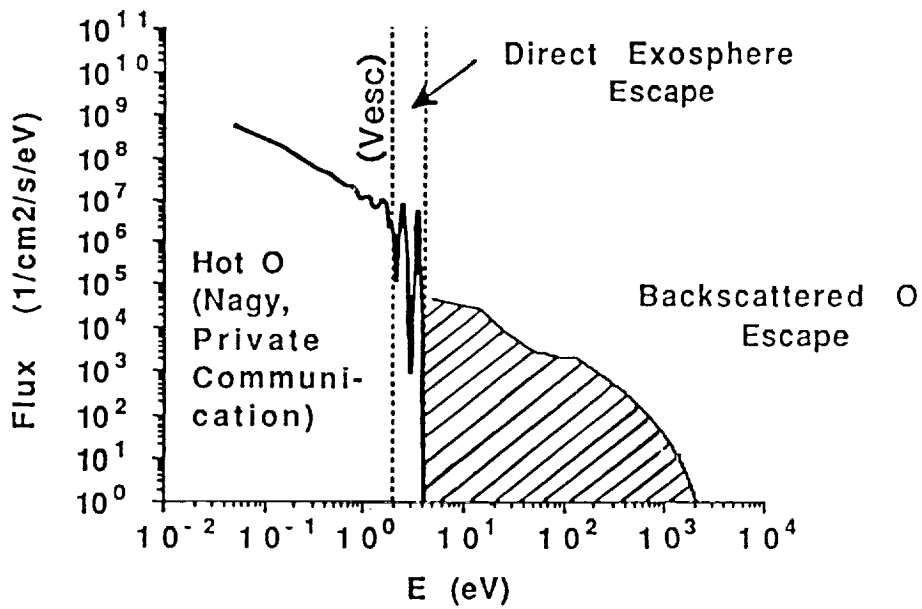


Vu-graph 12

Mars - Upward Moving O Fluxes

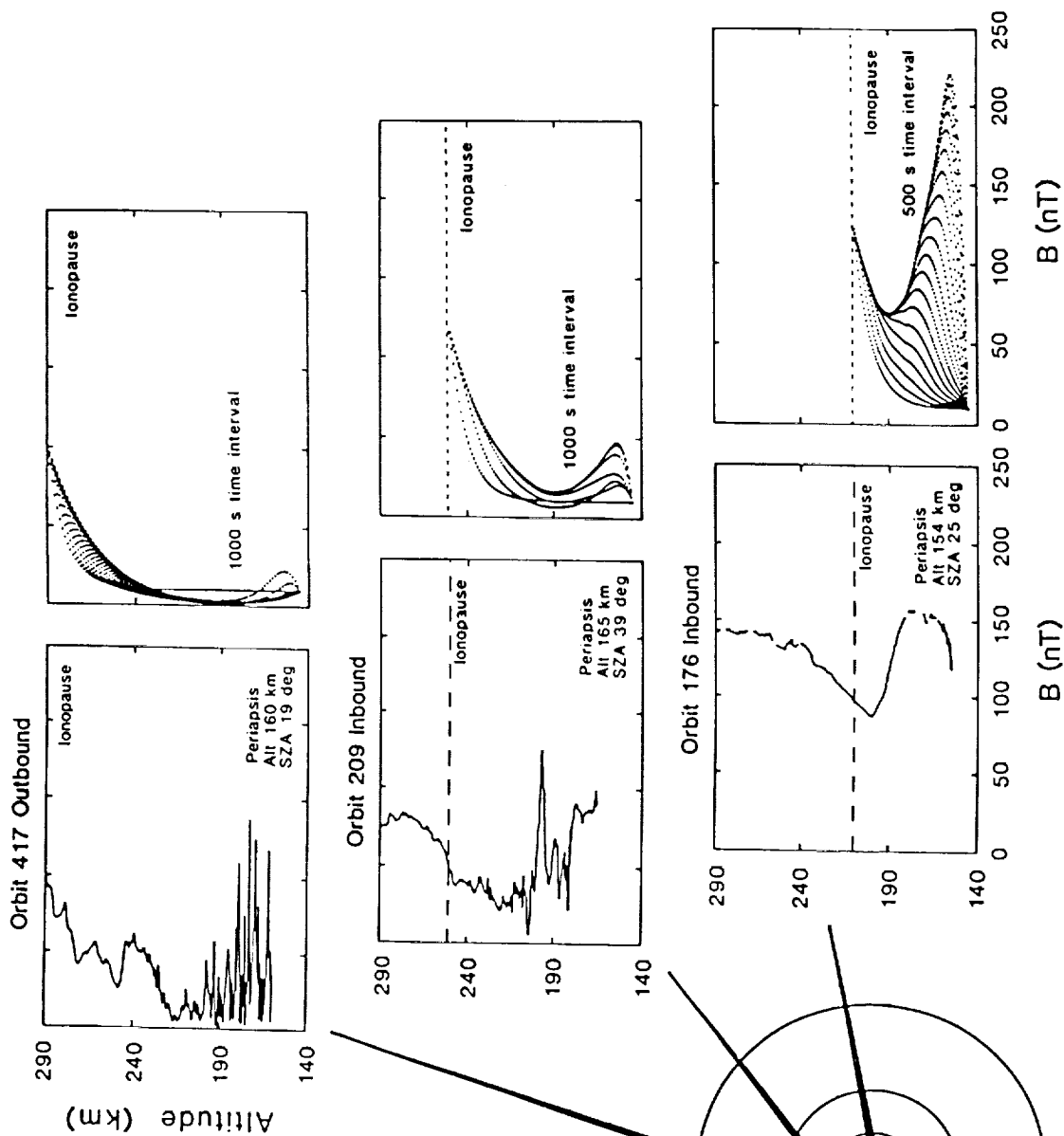


Mars - Upward Moving O Fluxes

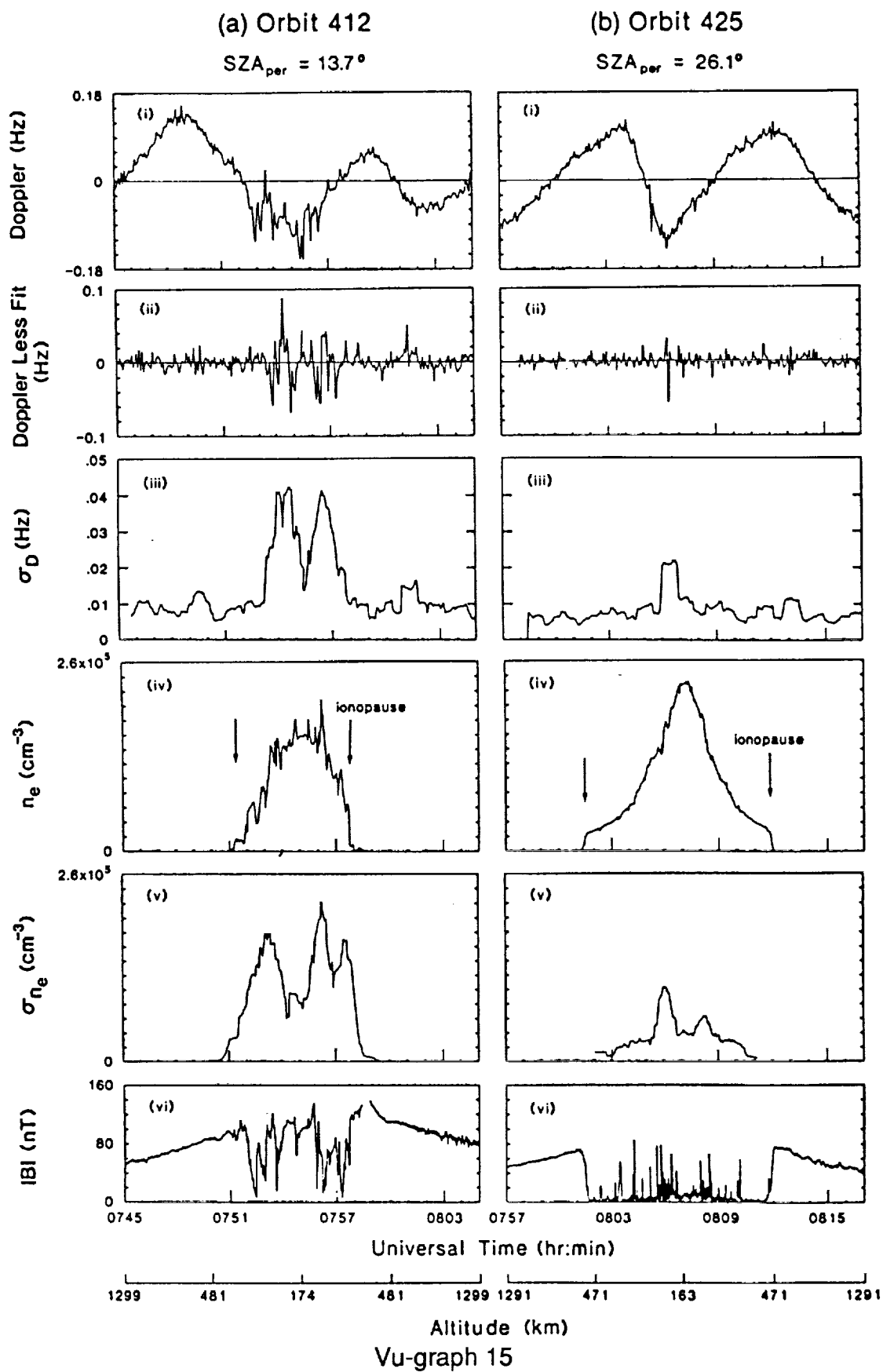


Vu-graph 13

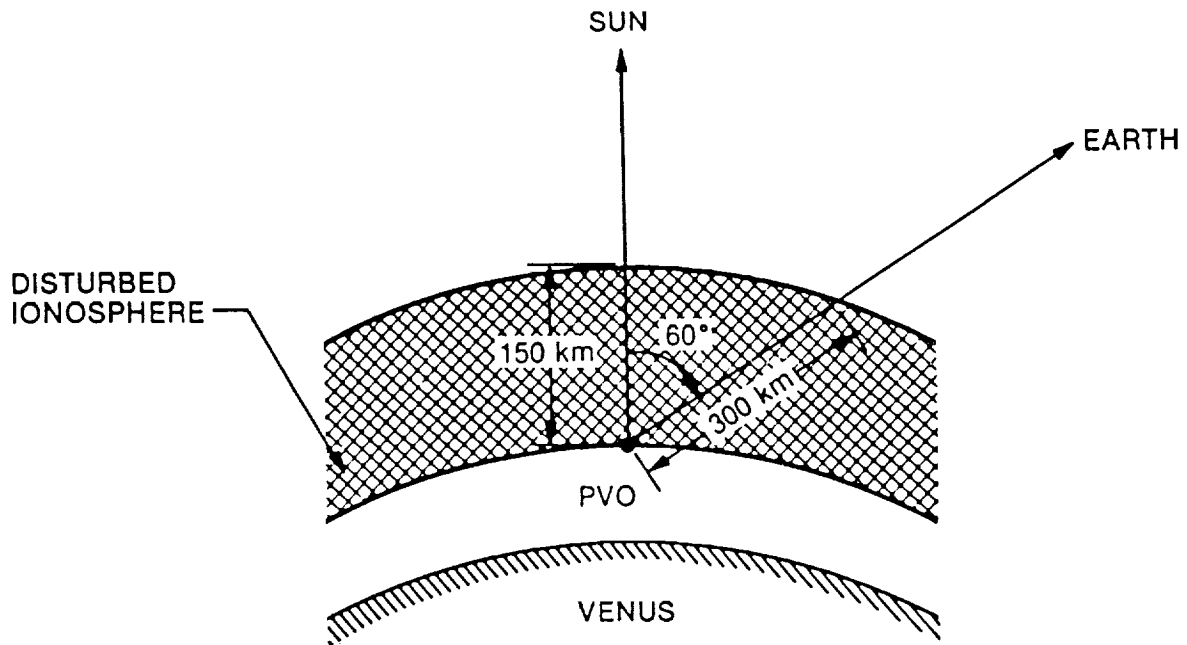
Pioneer Venus



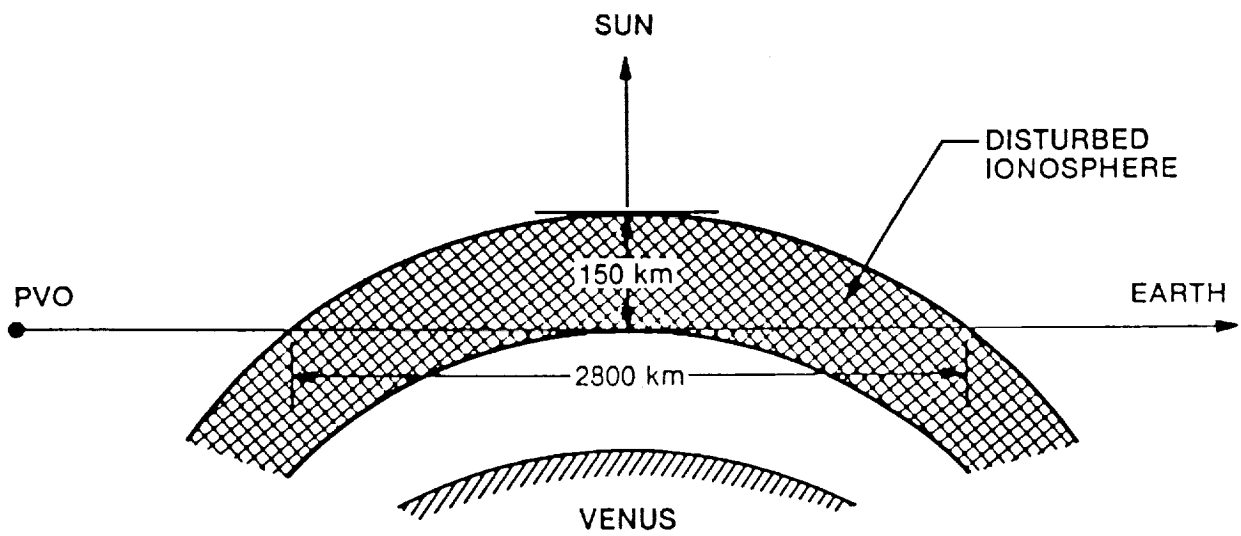
Vu-graph 14



(a)

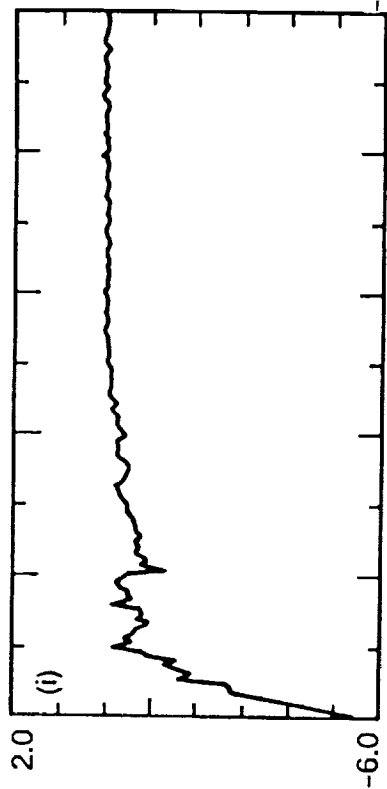


(b)

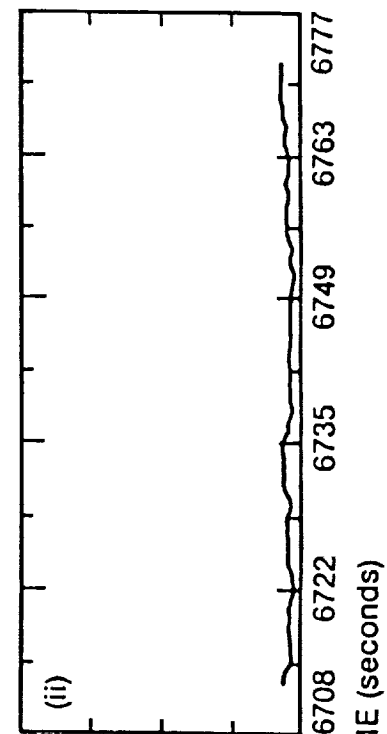
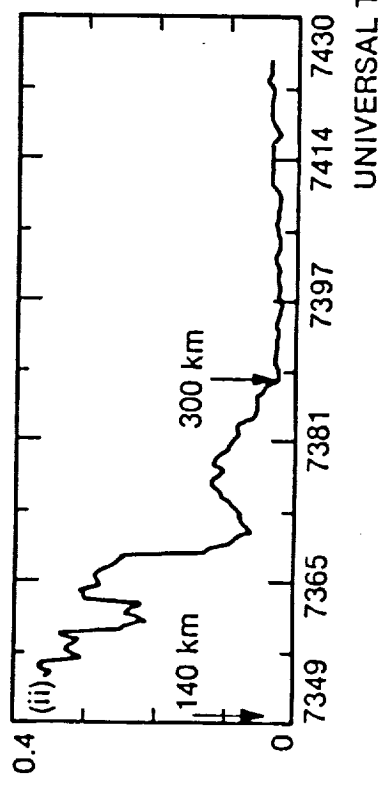
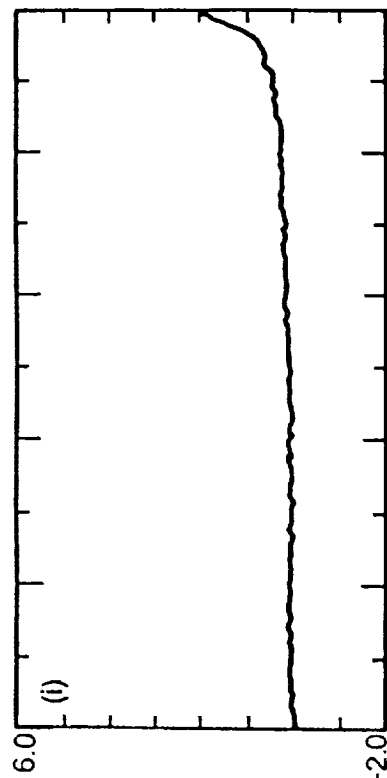


Vu-graph 16

(a) ORBIT 1053 EXIT
 SZA = 13.7°



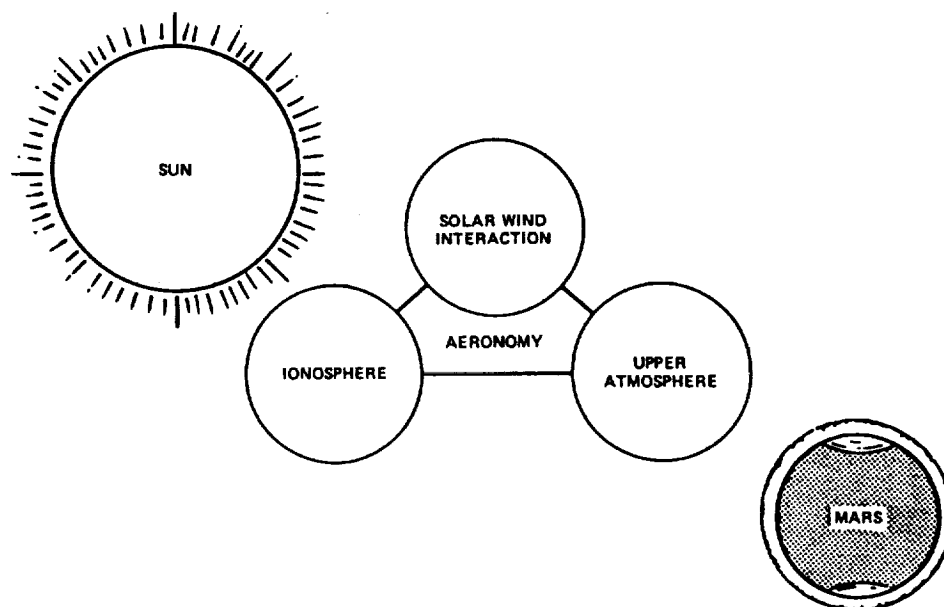
(b) ORBIT 1028 ENTRY
 SZA = 83.5°



UNIVERSAL TIME (seconds)

NASA Technical Memorandum 89202

Mars Aeronomy Observer: Report of the Science Working Team



October 1, 1986

NASA

National Aeronautics and
Space Administration

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

Vu-graph 18

TABLE V.1 MAO SWT RECOMMENDED INSTRUMENTS

	MASS	POWER	TELEMETRY ¹
<u>CORE PAYLOAD</u>			
Neutral Mass Spectrometer ² (NMS)	10.0	8.5	180
Fabry-Perot Interferometer (FPI)	13.5	5.5	30
UV + IR Spectrometer (UV + IRS)	5.0	7.0	130
Ion Mass Spectrometer (IMS)	2.5	1.5	60
Retarding Potential Analyzer + Ion Driftmeter (RPA + IDM)	4.5	4.0	80
Langmuir Probe (ETP)	2.0	4.0	30
Plasma + Energetic Particle Analyzer (PEPA)	10.0	9.0	320
Magnetometer (MAG)	3.0	3.5	200
Plasma Wave Analyzer (PWA)	5.5	3.5	130
Radio Science ³ (RS)	4.5	12.5 ⁴	-
	60.5 kg	59.0 W	1160 bps
<u>SECONDARY PAYLOAD</u>			
Infrared Atmospheric Sounder (IAS)	8.0	7.5	260
UV + Visual Synoptic Imager (UV + VSI)	9.0	8.0	1000
Neutral Winds/Temperature Spectrometer (NWTS)	10.0	9.0	180
	27.0 kg	24.5 W	1440 bps
TOTAL	87.5 kg	83.5 W	2600 bps

¹ Individual instrument rates can be highly variable and will depend upon the final payload and orbit selection. The rates listed are based upon typical duty cycles for each experiment and they have been averaged over the orbit (i.e., 6,000 x 150 km orbit has been assumed).

² Includes limited wind measuring capability.

³ Consists of S-band transponder and stable oscillator.

⁴ 10 W (continuous) for the stable oscillator and 25 W (10% duty cycle) for the S-band transponder.

CORE PAYLOAD

Neutral Mass Spectrometer	Number densities of neutral species, isotopic abundances, temperatures, and two components of cross-track wind velocity.
Fabry-Perot Interferometer	Atmospheric vector wind and temperature altitude profiles in the lower thermosphere ($h < 200$ km), metastable densities, volume emission rate profiles, rotational temperatures, and velocity distributions for escaping atomic oxygen.
UV+IR Spectrometer	H, O, C, and N altitude profiles from limb observations and nadir column densities of O ₃ and CO ₂ which allow for modeling of escape rates and surface densities; IR channel measures water vapor.
Ion Mass Spectrometer	Number densities of ion species and their isotopic abundances.
Retarding Potential Analyzer and Ion Driftmeter	Ion temperatures, densities, and velocities in the ionosphere.
Langmuir Probe	Electron temperature and density; solar EUV flux monitor.
Plasma and Energetic Particle Analyzer	Solar wind and magnetospheric particle velocity, density, temperature, and composition.
Magnetometer	Magnetic field properties in the solar wind, magnetosheath, magnetosphere, and ionosphere.
Plasma Wave Analyzer	Plasma wave properties in the solar wind, magnetosheath, magnetosphere, and ionosphere.
Radio Science	Atmospheric and ionospheric density and temperature altitude profiles.

SECONDARY PAYLOAD

Infrared Atmospheric Sounder	CO ₂ , H ₂ O, aerosols, thermal structure, and winds ($h \leq 70$ km).
UV + Visual Synoptic Imager	Global observations of stimulated emissions due to charged particle precipitation and NO, O ₃ , and dust in lower atmosphere.
Neutral Winds and Temperature Spectrometer	Thermospheric winds and temperature.

Vu-graph 20

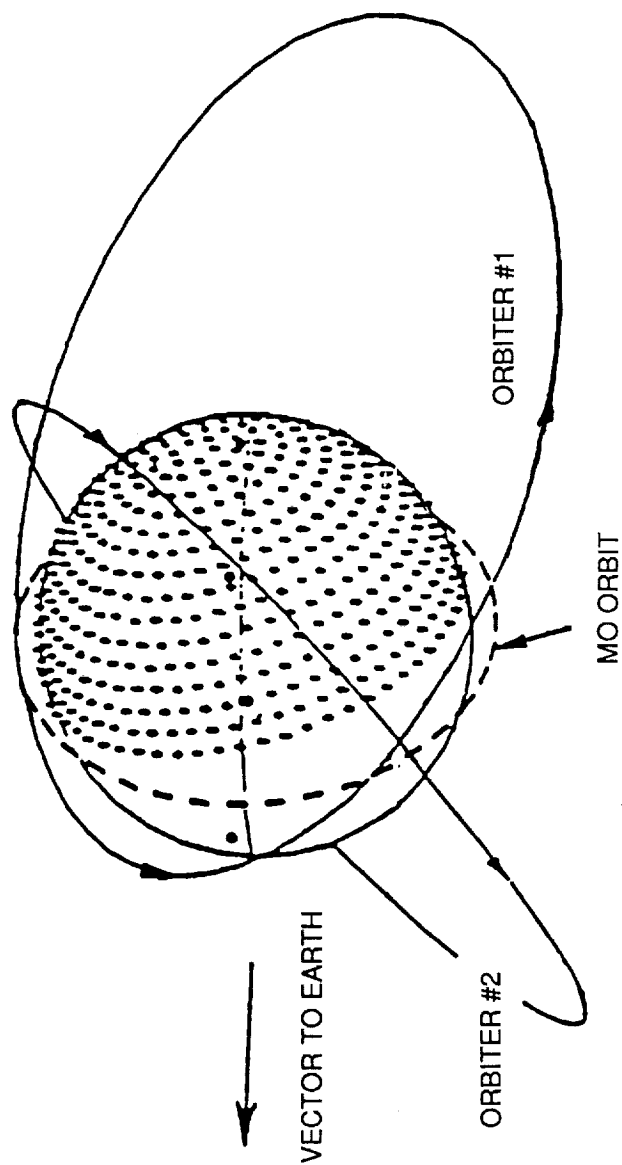
TABLE V.2 MAO SCIENCE OBJECTIVES AND INSTRUMENTS

Regions					
Neutral Atmosphere Structure		Ionospheric Structure		Solar Wind Interaction	
NMS		IMS		PEPA	
FPI		RPA + IDM		MAG	
RS		ETP		PWA	
UV + IRS		RS		IMS	
IAS*		MAG		RPA + IDM	
NWTS*		PWA		ETP	
UV + VSI*		PEPA		RS	
				UV + VSI*	
Processes					
<u>Photochemistry + Escape Rates</u>	<u>Global Energy Balance (Thermal+Winds)</u>	<u>Nightside Ionosphere</u>	<u>Dayside SW Interaction</u>	<u>Pickup Of Planetary Ions</u>	<u>Magnetotail Dynamics</u>
NMS	NMS	IMS	PEPA	PEPA	PEPA
IMS	FPI	RPA + IDM	MAG	MAG	MAG
UV + IRS	NWTS	ETP	PWA	PWA	PWA
ETP	ETP	PEPA	IMS		
IAS*	IMS	MAG	ETP		
NWTS*	RPA + IDM	PWA	RPA + IDM		
	PEPA	UV + VSI*			
	PWA				
	IAS				
	UV + VSI*				

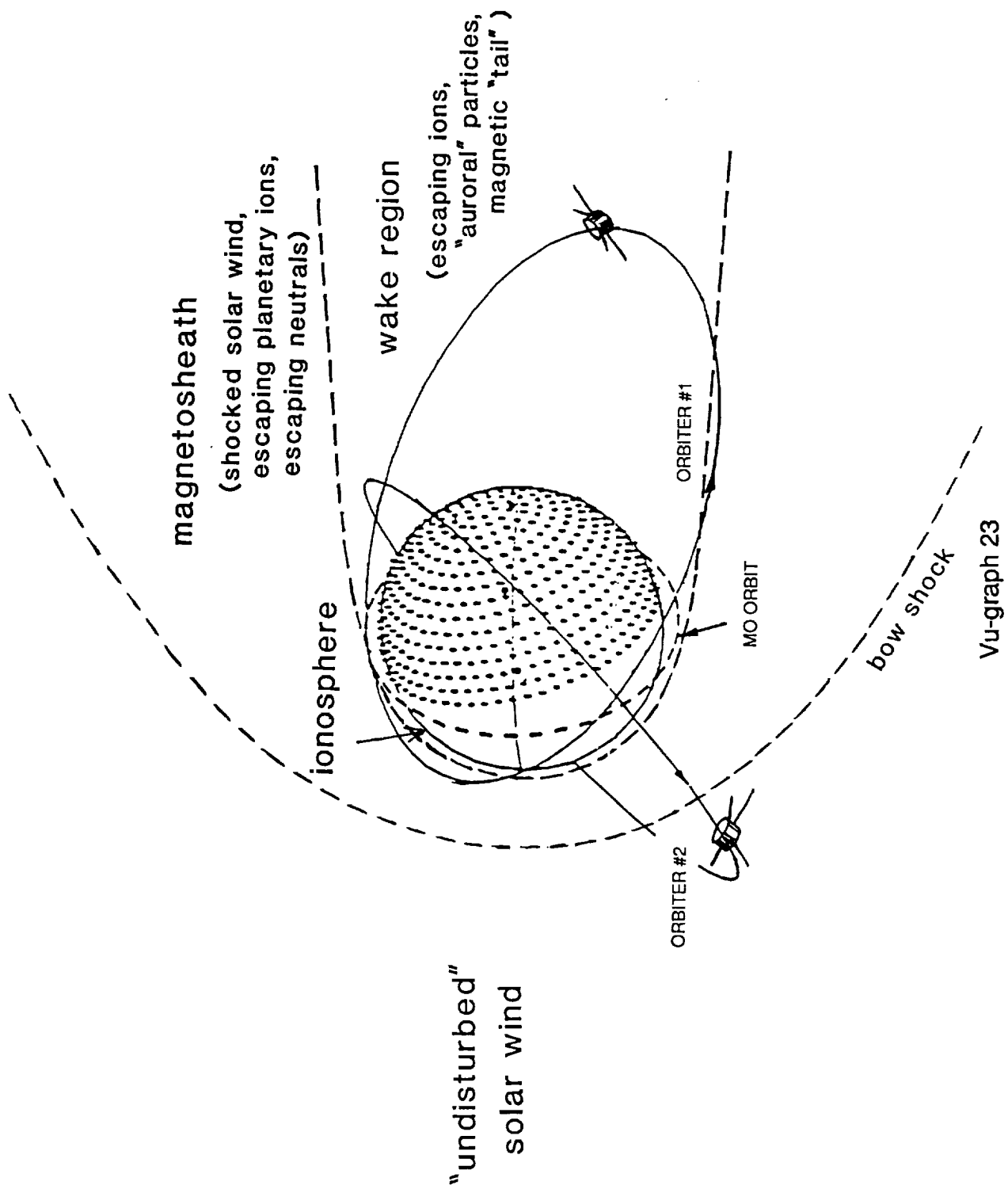
*Secondary payload instrument

Vu-graph 21

MARS NETWORK MISSION (candidate orbits)



Vu-graph 22



Vu-graph 23

Mars Aeronomy Mission

Desirable Characteristics:

Orbit:

elliptical (variable periapsis from <200 km (~ 110 - 150 km?), apoapsis ~ 1 - $5 R_M$)

polar (or high inclination)

rate of periapsis motion ~ 1 hour LT per week to sample all local times

spacecraft position in orbits can be controlled (phased)

Duration:

to cover as much of a Martian year as possible (for seasonal coverage)

Spacecraft:

Instruments identical on both spacecraft

onboard propulsion

body mounted solar cells (for low drag)

option to spin or despin (use momentum wheel)

extendable boom for magnetic, electric field experiments

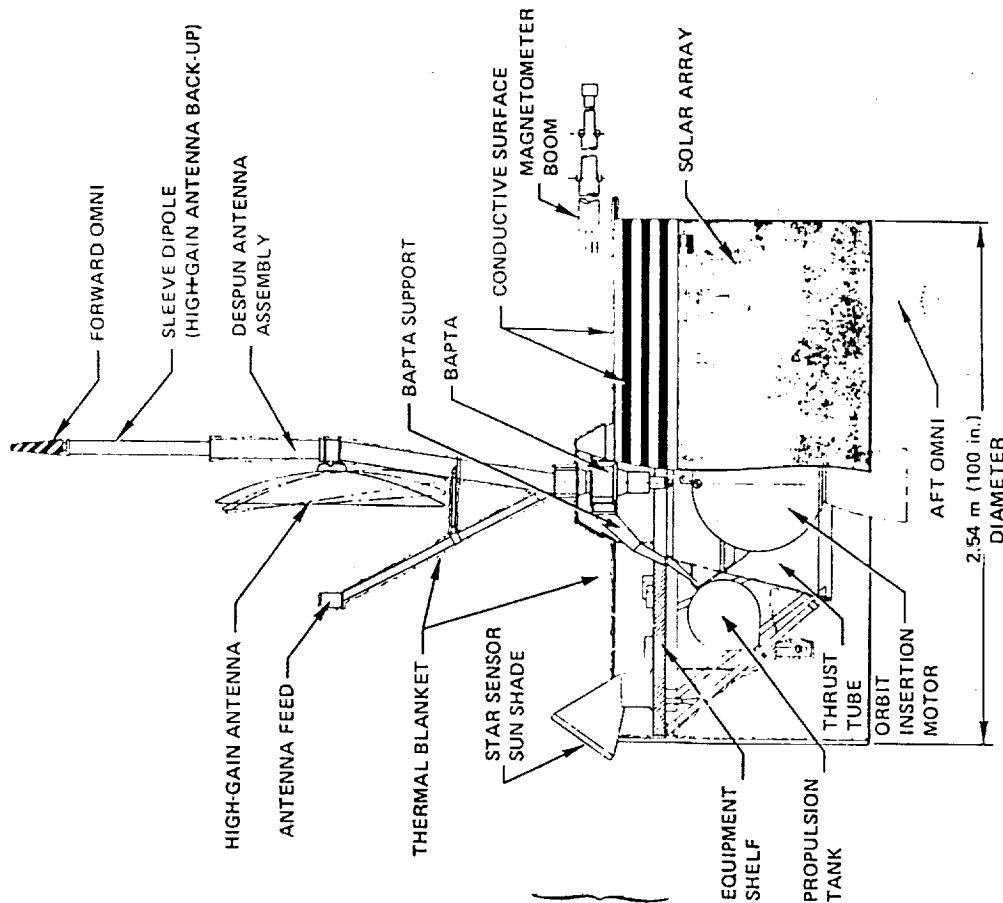
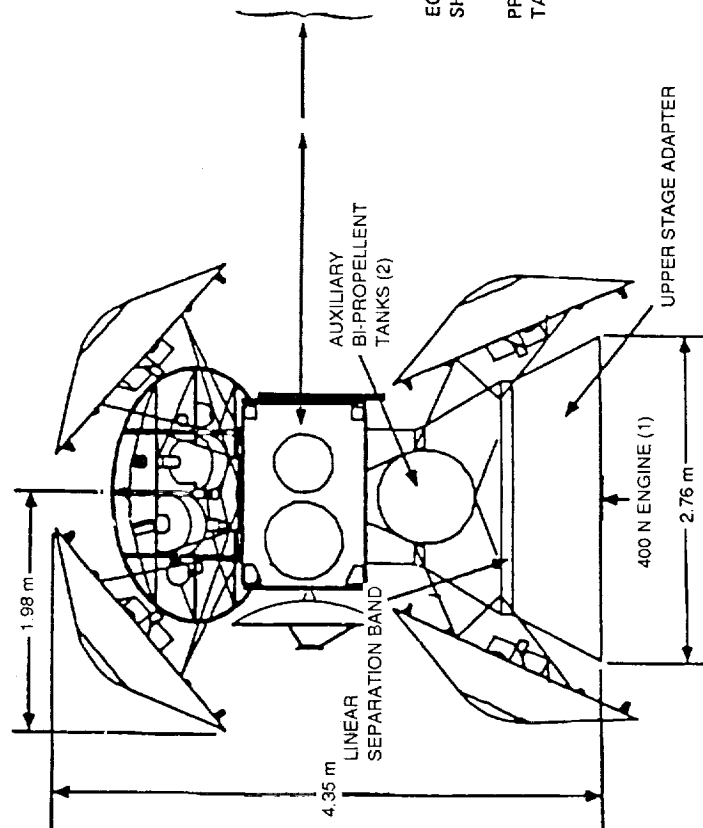
Periapsis Altitude versus Science

<u>Altitude</u>	<u>Science (cumulative)</u>
> 400 km	<ul style="list-style-type: none">- extended nightside ionosphere "rays"- escaping ions and neutrals- bow shock, magnetosheath, magnetotail- "PHOBOS-2" science
> 300 km	<ul style="list-style-type: none">- top of dayside ionosphere- precipitating ions- upper atmosphere
> 200 km	<ul style="list-style-type: none">- upper ionosphere (structure, dynamics, induced magnetic field)
> 150 km	<ul style="list-style-type: none">- intrinsic magnetic fields (?)
< 150 km	<ul style="list-style-type: none">- ionosphere peak (~120-130 km)- intrinsic magnetic fields (?)
< 100 km	<ul style="list-style-type: none">- aerobraking environment

Vu-graph 25

MARS NETWORK SPACECRAFT

STOWED CONFIGURATION
(ONE AEROSHELL REMOVED FOR CLARITY)

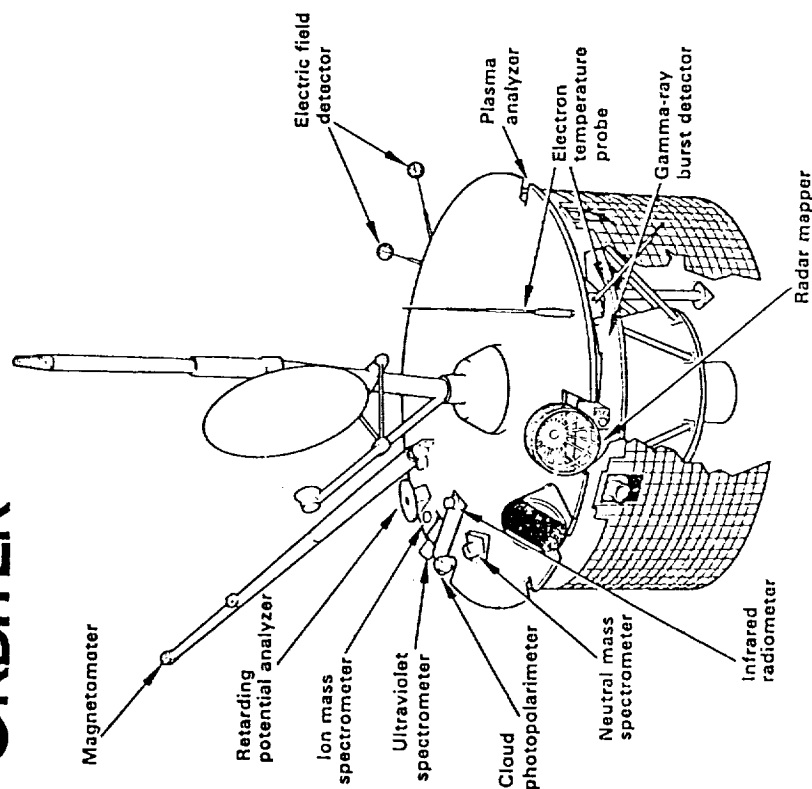


Pioneer Venus Orbiter

Vu-graph 26

ORIGINAL PAGE IS
OF POOR QUALITY

PIONEER VENUS ORBITER



The Orbiter spacecraft carries 12 scientific instruments. They will measure characteristics of the upper atmosphere of Venus in situ and characteristics of its clouds remotely. A radar mapper to study the surface of Venus is also onboard. The fields and particles experiments operate both en route and while in orbit around Venus. The actual communication signals will be used to make radio propagation measurements once the spacecraft is in Venus orbit. An X-band transmitter was added to the basic S-band link for this propagation research.

SCIENTIFIC INSTRUMENTS

Instrument/Principal Investigator	Objective
ORBITER	
Ion mass spectrometer H Taylor, GSFC	Measure composition of ionosphere and charged particle concentrations
Neutral mass spectrometer H Niemann, GSFC	Measure composition of upper atmosphere and neutral particle concentrations
Electron temperature probe L Brace, GSFC	Measure electron temperature and density profiles of ionosphere
Magnetometer C Russell, UCLA	Study interplanetary magnetic fields and Venus' magnetic field from lower ionosphere to solar wind
Plasma analyzer J Wolfe, ARC	Investigate interaction of solar wind with ionosphere
Electric field detector F Scarf, TRW	Map solar wind bow shock; determine interaction of solar wind and ionosphere
Gamma-ray burst detector W Evans, LASL	Measure gamma-ray emanations from outer space; correlate with sensors in other locations for pinpointing of sources
Retarding potential analyzer W Knudsen, LMSC	Investigate physics and ion chemistry of ionosphere
Ultraviolet spectrometer A Stewart, U of Colorado (CUVUS)	Determine energy balance of thermosphere; ionization rates, and O, CO, and CO ₂ composition
Cloud photopolarimeter J Hansen, GISS (OCPP)	Determine polarimetry of planet on global scale; pictorial mapping
Infrared radiometer F Taylor, JPL (OIR)	Investigate thermal structure of lower atmosphere and vertical distribution of particulate matter
S/X-band RF occultation RF Science Team	Derive temperature and pressure in lower atmosphere (to ~35 km); map electron profiles; determine neutral atmosphere density distribution and vertical cloud structure
Radar mapper Radar Mapper Team	Determine topography and electrical properties of surface

HUGHES

HUGHES AIRCRAFT COMPANY

Vu-graph 27

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- Woo, R., W. L. Sjogren, J. G. Luhmann, A. J. Kliore and L. H. Brace, Solar wind interaction with the ionosphere of Venus inferred from radio scintillation measurements, *J. Geophys. Res.*, 94, 1473, 1989.

Session A, Submittal No. 10

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Specification Goals for a Mars Seismic Network

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Abstract

A seismic network on Mars should: 1) have enough stations (e.g., 24) to characterize the seismicity of the planet for comparison with a diversity of structural features; 2) be comprised of low noise stations, preferably underground, 3 to 4 orders of magnitude more sensitive than those used on Viking; 3) record over a sufficient band-width (DC-30 Hz) to detect micro-earthquakes to normal modes; 4) record for a sufficient duration (10 years) and data rate (10^8 Mb/day/station) to obtain a data set comparable to that from the Apollo mission to the Moon so that locations of major internal boundaries can be inferred, such as those in the Earth, i.e., crust - lithosphere - asthenosphere - upper - lower phase transitions - outer - inner core. The proposed Mars Global Network Mission provides an opportunity to sense the dynamics and probe the interior of the planet. We discuss the seismic objectives, the availability of the instrumentation and trade-offs to meet them.

Introduction

The science objectives of the Mars Global Network Mission include installation of a seismic network on Mars in order to measure the seismic activity of the planet and to characterize its structure for comparison with Earth. Preliminary specifications for the mission call for installation of up to 24 penetrators or hard-landers on Mars, in pairs, at 12 widely dispersed locations. Landers making up each pair will be installed hundreds of meters to several kilometers apart, thereby achieving some redundancy. We review here the science objectives of the seismic experiment, the instrumentation specifications required to meet these objectives, and report on some recent progress on construction and testing of a prototypical hard-lander seismometer.

Science Rationale for Seismic Network on Mars

Seismology has told us more about the Earth's interior than any other geophysical method. Such information from Mars is vital to progress in understanding the evolution of the solar system. The Viking spacecraft landed on Mars in 1976. The seismometer on Lander I failed to uncage whereas that on Lander II provided 0.24 Earth years of observational data (Goins and Lazarewicz, 1979). The Lander II data contained mainly wind

noise and possibly one marsquake but even that is doubtful. The seismic part of this mission was of secondary importance to the search for life experiments. We are not yet sure that marsquakes exist.

Apart from the uncaging problem on Viking I, wind noise on Viking II was extreme because the instrument was located high up on the Lander near antennae, which vibrated or rocked the structure in response to the wind forces. Also, because only one instrument operated on Mars, it was almost impossible to tell if a given event was a wind burst or a marsquake. The seismometer was less sensitive than the Lunar (Apollo) instruments due to size, weight and power constraints. However the experiment did place bounds on noise levels. It has been estimated that a network of "seismometers more sensitive than the Viking instrument by at least a factor of 10^3 "... "emplaced by penetrators or deployed as small packages can operate on the planet without being affected by typical Martian winds" (Anderson et al., 1977).

Science Goals of Mars Seismic Network

Scientific questions that a seismic network on Mars can address depend on whether the instruments are short period (10 seconds to 10 Hz) long period (DC to 10 seconds) or broad-band (DC to 30 Hz) and whether they are 1-component or 3-component. Ideally they should be 3-component, broad-band, but this places severe constraints on installation, and volume and weight of the instrument package, but has the return that the science goals will be met faster than if the performance is restricted. Table 1 lists the seismic science goals separated into those achievable with short period instruments and long period instruments.

Short Period Seismometers

1. Are there marsquakes?
2. How do their locations compare to structural features such as rift zones, volcanoes, and uplift zones?
3. How does the attenuation of seismic waves compare with Earth and the Moon where an order of magnitude difference was observed?
4. Are there major internal boundaries in Mars similar to those within Earth and the Moon, i.e., crust-lithosphere-asthenosphere-upper-lower phase transitions-outer-inner core?
5. Is there sub-surface structure that yields information on the Martian hemispheric dichotomy (e.g. 1=1 convection)?
6. What are the dynamics of impacts on Mars from meteorites?

7. What are the focal mechanisms of marsquakes and how do they relate to inferred stress fields, e.g., from isostatic imbalance?

Long Period Seismometers

8. Do large impact events or marsquakes generate measurable normal modes which can be used to estimate velocity and density distribution?

9. Can we detect surface wave dispersion?

10. What is the Love number of the Planet?

11. Can we detect annual or Chandler wobble generated by internal changes of the moment of inertia?

Table 1 Scientific Questions for Mars Seismic Array

Science Goals of Mars Seismic Network

If we knew Mars as well as we know the internal structure of the Earth from seismology, not only would we be exploring a new planet, we would also be adding fundamentally to our understanding of the evolution of the Solar System including the formation and composition of both Mars and Earth. Solar Nebular theories of the compositions of the planets predict that the volatile content, oxidation state and silicate iron ratios increase with distance from the Sun. The distribution of elements within a planet is determined by the temperatures during formation. For Mars we know only the mean density and moment of inertia (and there is still considerable debate on this, Kaula et al., 1989, Bills, 1989). Further progress is hampered because models satisfying these constraints allow trade-off between mantle and core densities, and core size. Direct determination of the size of the core and density profiles, by seismic means, would constrain the overall composition of the planet. Models of the thermal evolution of Mars (Schubert et al., 1989) since formation differ as to whether the core is solid or molten. An important factor in this regard is the amount of Sulphur in the core, which if it is the 15% as inferred from the SNC meteorites, results in a completely molten core, but if much less, can result in a solid core. Attenuation of S-waves would tell us about the fluidity of the core.

We assume that Earth's core is mainly iron but with a substantial amount of lighter element, or elements, based on estimates of uncompressed density, shock wave data, and consistency with meteorite (type I carbonaceous) compositions. There are nonetheless uncertainties associated with this view. Are the finite strain theories used for decompression of the density truly applicable? What is the light element, or elements? Are the meteorites a relevant geochemical reference frame? Comparison of Earth with another planet will allow us to test the hypotheses used on Earth.

Installation of Mars Seismic Network

Various methods to install seismometers on Mars include implantation by penetrator, deployment on the surface from a rover, or by hard, rough or soft-landers. The g loads on the instrumentation range from thousands of g for a penetrator and hard lander, hundreds of g for a rough lander and tens of g for a soft lander.

Penetrators

Penetrators offer an attractive way to implant a seismometer because the seismometer is firmly coupled to the planet, and is unlikely to experience the wind generated rocking motions that were thought to have generated noise on the Viking instrument (Anderson et al., 1977). Penetrator technology is well advanced. Approximately 18,000 penetrators were dropped in Southeast Asia and radioed information on troop movements from seismic and microphone sensors which was detected by planes at 20,000 feet. The idea of a penetrator mission to Mars dates back to reports by JPL (Briggs et al., 1975) and Sandia (Lumpkin et al., 1974). Other studies made in the mid seventies include those by Westphal et al., 1976, Blanchard et al., 1976, and Greely and Bunch, 1976.

Burial of the seismometer beneath the surface by a penetrator will reduce wind noise. Also remoteness from a lander will eliminate internally generated spacecraft noise, both electrical and mechanical, as well as wind generated vibrations of the superstructure. Burial will also keep the seismometer thermally insulated from diurnal and other surface temperature changes. This is critical for long period seismometers which, if installed at the surface, record strong signals generated by thermoelastic strains, both in the surrounding rock and in the instrument itself. At short periods, thermoelastic changes are buffered by the thermal inertia of the instrument.

Presently we expend much effort digging pits to install sensors 1.5 m into the ground in our field installations on Earth. For short period recording, it suffices to cover the pits. For intermediate period recording, the pits are filled with insulation. However, first class seismic observatories are usually located in vaults deep underground such as mine shafts, tunnels, or in bore-holes. A penetrator installation on Mars is a practical compromise.

Surface Versus Penetrator Installation

A surface installation, though attractive because of its simplicity, compromises the quality of the seismic data obtainable. Ground coupling can not be assured. Proximity to wind and temperature changes would probably limit the instrumentation to short period only. However surface installations worked on the Moon, though they did not have to deal with winds. There are, however, advantages to designing two types of landers, a surface one for the seismic package and a sub-surface penetrator for short-lived (1 month) experiments such as soil properties, mass spectrometry etc. It would remove the need for a small RTG, since the short term experiment in the penetrator could run on lithium batteries. It also

removes the possibility of contamination of the chemical analyses by radioactive products from the RTG. A softer surface lander for the seismometer would reduce the shock tolerance requirements for the RTG and seismometer system. This may be critical for the RTG since, because of its extreme temperature (1000°C) the thermocouples can not be bonded; it may not survive shocks greater than a few hundred g.

Although it should be tested, it is probable that a large proportion of seismic short period information on Mars could come from instruments installed on the surface. The trade-off in simplifying the installation would be the loss of long period signals. Also, the low end of the short period band would be noisier than that at depth. We ran a series of tests in the alluvium in the caldera at Long Valley, California, in which a short period sensor was buried and the background noise measured as a function of depth in a wind of about (4.0 m/s) 8 knots. In the frequency band tested, 5 Hz - 30 Hz, there was no perceptible difference in background noise. Such tests need to be performed over the full frequency range and for different wind and surface conditions, before effects of burial can be quantified. Shedding wind vortices from obstacles can generate noise in the seismic band dependent on wind speed and obstacle shape.

Viking mission data showed that mean seismic amplitude increased as the wind velocity squared (Anderson et al., 1977) for winds ranging from 3 m/s to about 10 m/s. Optimal design of a surface installation will require the instrument package to be of a streamlined shape. It will need to have the capability to attach to the surface securely. It will also need to be kept isothermal (gradients less than 10^{-5}C/m) and at constant pressure (to within 10 mbar).

Table 1 shows the science objectives (1-7) that could be achieved with a short period seismometer installed at the surface. We could measure the seismicity, the travel times, fault plane solutions, invert travel times for radial structure, including detection of the Martian core. We would miss out on (8 - 11), in particular, surface waves and normal modes, which would be regarded by most seismologists as an extremely high price to pay.

Normal modes will give an independent check on the radial structure determined from travel time analysis of body waves. One large marsquake which generated a wide spectrum of normal modes would allow inversion for internal radial structure; that would take years using short period travel-time data alone. Measurement of lateral variation in the excited modes, at multiple stations, can be used for determination of global heterogeneity. Surface waves measured at multiple stations provides a method to measure upper mantle lateral heterogeneity, which will be particularly interesting beneath the Tharsis plateau region.

Detection of lateral heterogeneity means all stations should be broad-band. We conclude that too much science is lost if the seismic installations are restricted to (surface) short period installations. All instruments should be broad-band, installed either in penetrators at depth or, if on the surface, they should have good coupling, preferably to bedrock, and be insulated from temperature and pressure fluctuations.

Data Acquisition Specifications

Mars' seismic activity is thought to lie between that of Earth and that of the Moon (Kaula, 1984). If Mars' seismicity obeys a Gutenberg Richter law, with b value = 1, such as is observed on Earth, with instruments a factor of 10^3 more sensitive than Viking, 3 orders of magnitude more earthquakes should be detectable. As well as marsquakes, landslides of over steepened crater walls and meteorite impacts will generate seismic signals. On Earth, installations of comparable sensitivity to that proposed for Mars, detect about 1 earthquake of magnitude=5.5 per day world-wide. A marsquake of this magnitude would probably not have been observable had it occurred further than 90 degrees from the Viking instrument.

If Mars seismograms are similar to those on Earth in order to capture the important phases, P,S, ...multiple ScS etc., recording at 50 samples/second should continue for several hours after initiation of a moderate sized event. After this time a low sampling rate (1 sample/second) could be used to detect normal modes. In areas of seismic swarm activity, for example active volcanic regions, the local earthquake activity can be as much as 100 events per day, requiring continuous recording.

On the Moon, an average of 4 events per day were detected comprised of: unclassified events (2.4/day), deep moonquakes (1/day), meteoroid impacts (0.6/day). Events on the Moon persisted for several hours, because of the high Q (4000) of the Lunar mantle (Dainty et al., 1976; Nakamura et al., 1976). For a Lunar-type activity it would be necessary to save data for several hours per day, at 50 samples/second, to record the full wave trains of the seismic signals.

These considerations indicate that the daily data budget of a seismic station can be calculated as 3 components at 50 samples per second for 24x3600 seconds at 30 bits per sample (24 bit A/D and 6 bit gain range) = 3.88×10^8 bits/day. With data compression, such as event detection, this number can be reduced; 10^8 bits/day per station would provide an adequate coverage. If 2 transmissions were made to an orbiter per day, this amount of data would require an on-board 6 Megabytes of RAM.

Investigation of seismicity requires setting up a network of at least 3 stations since this is the minimum needed to locate an event. However to measure local, regional and global seismicity at least 9 should be installed, that is, a 3-station local network with stations separated by about 20 km, a regional network of separation 200 km and three stations distributed across the planet. We propose that the local array be installed in the Tharsis region where earthquakes are expected from the associated stresses due to inferred isostatic imbalance. The regional and global networks would extend out from this base. To measure seismicity at diverse structural settings, several local networks should be installed. The proposed network of 24 seismometers at 12 different locations with closely separated pairs will achieve these goals.

Seismometer Specifications

There is currently no seismometer available that would withstand the shock associated with penetration. Either presently available ones, with the desired sensitivity, will have to be modified, or a new design implemented. The seismometer design should be predicated on considerations of ruggedness and simplicity. Leaf spring seismometers such as the Ranger (Kinometrics, Pasadena, California) have the required ruggedness.

In 1962 Lehner et al., (1962) report (2000') drop tests from a helicopter of the Ranger seismometer which was clamped with all moving parts immersed in fluid (150 cc of n-heptane). Decelerations were in the range 3000-7000 g. After cushioning the various components, the final design survived a series of 7 drops with no degradation of performance.

Coil spring designs such as the Mark products (Houston, Texas) L4C or the HS10 (Geospace, Texas), are also rugged but have less tolerance to non-verticality. The response of a damped inertial seismometer depends on the mass, the spring constant and the damping factor. The low frequency response of a velocity transducer is critically dependent on the value of the resonant frequency. Since the response to ground displacement falls off as about $1/(\text{frequency squared})$ the useful bandwidth is about a decade above and below resonance. With high signal to noise ratio and wide dynamic range, the useful bandwidth can be extended to 3 decades, e.g., 0.01 Hz to 10 Hz, for seismometers of resonant frequency 0.5 Hz. However a typical range for an L4C, as used in the USGS network in Southern California, is 0.1 to 10 Hz.

One way to extend the dynamic range and linearity of an inertial seismometer is to use force-balance feedback in the form of either a magnetic or electrostatic restoring force proportional to the ground acceleration. The former consumes power whereas the latter, while consuming negligible power, provides a weak force and is typically used on long period instruments (such as the LaCoste gravimeters of the IDA array). Alternatively addition of a displacement transducer, sensitive to sub angstrom displacements, can provide a low frequency channel output with flat response to ground acceleration with a minimal power requirement.

The final position of the penetrator may be well off vertical. The seismometer must either work at any angle or have a levelling mechanism. Seismometers with the mass suspended from coil springs have little clearance and so jam if they are not close to vertical. For example, the L4C jams at 17° off vertical. The mass of the Ranger seismometer is attached to leaf springs at either end so that when it is tilted the transverse shear strength of the flat springs prevents lateral movement which would otherwise cause it to jam against the casing. In fact it can be converted to a horizontal seismometer merely by rezeroing the mass to the position of greatest sensitivity. The commercially available Ranger from Kinometrics has a diameter of 11.1 cm excluding casing. This is too large to be directly transferred into a penetrator (diameter 9 cm). A seismic sensor is required that has the versatility and ruggedness of the Ranger but is small enough to fit in a penetrator and has a broad-band transducer.

In 1977 the Bendix Corporation (Perkins 1977) presented a design for a multi-instrument penetrator including a 3-axis seismometer. The seismometer was attached to a levelling table pivotted on a monoball bearing. Two motor drives 90 degrees apart, attached to the table via spherical bearings and flexures, are used to level to within 1 microradian. The transducer consists of a vertical geophone and a North American Rockwell biaxial bubble tiltmeter which can both be used as a two axes horizontal seismometer and also as the levelling transducer. However this apparatus was not built. Levelling to within 10^{-6} is difficult. We favor a simpler design which does not have such stringent levelling requirements.

Number of sites	12
Number of landers	24
Number of channels/lander	3
Number of samples/s	50 sp/s
Bandwidth	DC-30 Hz
Sensitivity	$10^{-11}g$
Displacement Resolution	$10^{-12}m$
Free period	0.6 seconds
Power	100 milliwatts (sensor) 1.0 watts (signal acquisition)
Data rate	1000 b/s or 10^8 b/day
A/D	24 bit
Gain Ranging	6 bit
Dynamic Range	140 dB
Clip Level	$10^{-4}g$
On Board RAM	6 Megabyte
Clock Accuracy	10 millisec
Calibration	1/day
Shock	10,000g for 2 millisecs any axis
Weight seismic mass	3x0.16 kg
Weight sensor	
(exclusive of housing)	3x0.5 kg
Diameter	9.0 cm
Height	3x15.0 cm
Temperature control	$10^{-5}^{\circ}C/m$
Vacuum	10 mbar
Spring resonances	≥ 80 Hz
Mass Centering	+ - 90 degrees; 6 volt motor

Table 2.
Specifications for Mars Global Seismic Network

Specifications for the Mars Network Seismic stations are listed in table 2. Seismometer specifications are based on presently available force balance seismometers, including the

Guralp (Guralp Systems, Reading, England) seismometer and the Strekeisen seismometer (Wielandt and Strekeisen, 1982) which have the sensitivity required but, owing to the Bendix hinges that support the boom, they do not have the required ruggedness. Specification of the digital acquisition system is based on systems currently in use by IRIS (Incorporated Research Institutions for Seismology) for the permanent and portable networks.

Brassboard Prototypical Penetrator Seismometer

One of the most popular modern broad-band seismometers is the recently developed Guralp force-balance feedback seismometer, the mechanical part of which resembles, in many ways, a leaf spring micro-gravimeter designed by R.V. Jones (Jones and Richards, 1973). The difference is that the Guralp employs Bendix hinges to pivot the boom with a leaf spring supplying a restoring torque whereas in the R.V. Jones design, the leaf springs also perform the function of the hinge. The Bendix hinges are too weak to withstand the high deceleration impacts.

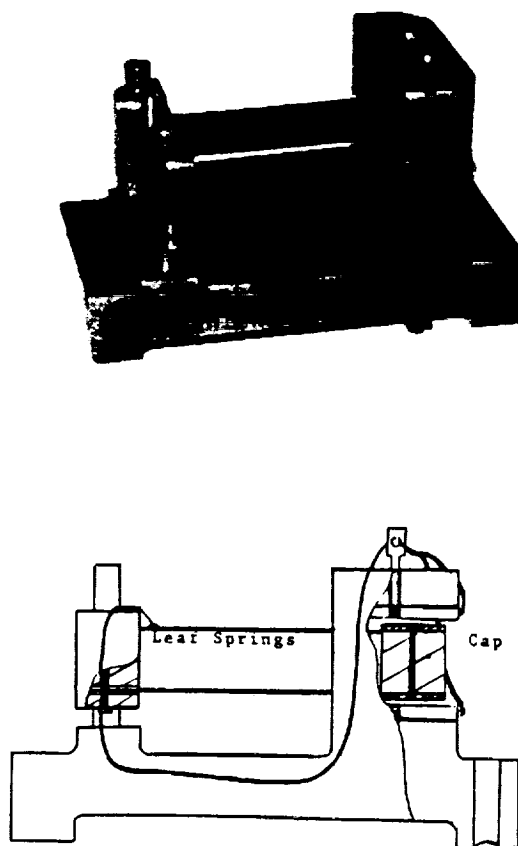


Figure 1 Leaf spring seismometer designed to be shock tolerant.

We have constructed a leaf spring seismometer based on the R.V. Jones design. This design has an advantage that it works 13° off-vertical without post-implantation adjustment and fitted with an adjustable re-zeroing mechanism would work in any orientation. Therefore a three component set could be installed in a penetrator for which the default would be no post impact adjustment, if the penetrator ends up close to vertical, and minimal rezeroing adjustment if it ends up well off vertical. Even then, if the rezeroing system fails, some data would be achieved, albeit at reduced sensitivity. Basically the ruggedness of leaf springs is achieved by employing 2 parallel Beryllium Copper springs on which the mass is suspended. A photograph and schematic of our sensor is shown in figure 1. Although it is more rugged than the Guralp seismometer, the trade-off is that it is about $1/3$ as sensitive.

The position of the mass is detected by capacitance micrometry. Eventually a magnet-coil assembly will be used to provide force feedback as in the Guralp seismometer. By adjusting the filters for the force feedback output a wide dynamic range can be achieved.

Implementation of a Laboratory Impact Tester

In order to test the prototype, we assembled a laboratory impact simulator (Kewitsch, 1989). This has enabled us to conduct impact tests in the laboratory at UCLA to eliminate obvious design flaws before going to the more extensive testing at Sandia National Laboratories, Albuquerque, or from helicopter drops. Validyne Engineering (Chatsworth, California) donated a drop tower to the project. We added 8 bungee cords stretched over a pulley system, allowing 100% stretch of the cords to accelerate the drop, to give an effective drop of 40 feet (figures 2 and 3). An accelerometer/charge amplifier system measures the deceleration; the output is recorded on a signal analyzer (see figures 2,3,4). The system was calibrated at Environmental Associates, Chatsworth.

IMPACT DEVICE

from top to bottom: 8,6,4 bungee cords

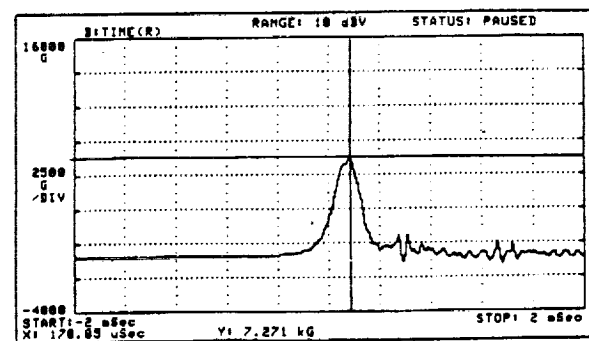
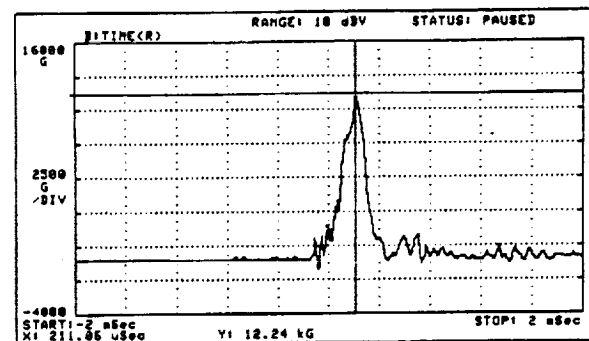
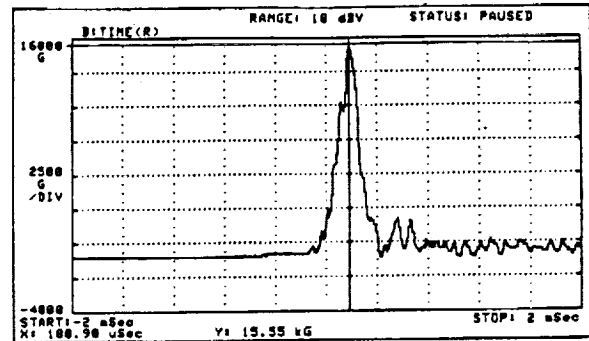
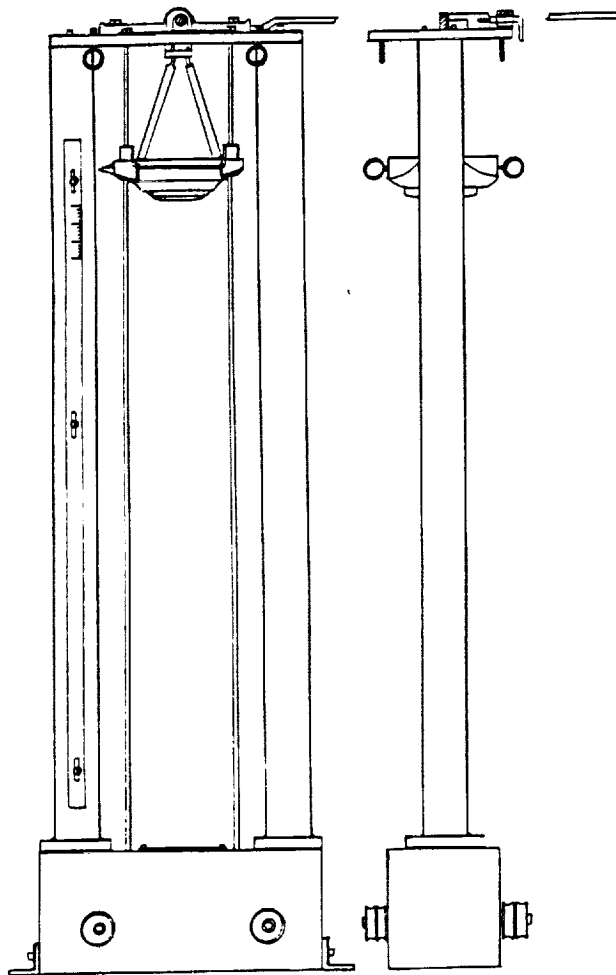
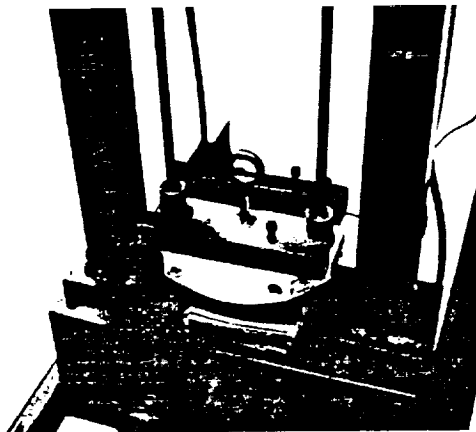


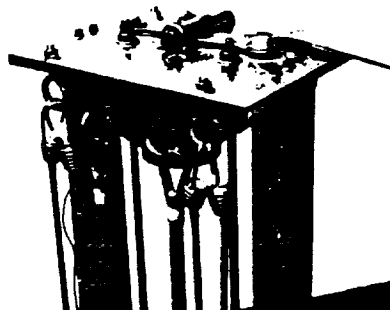
Figure 2,3 Schematic of Bungee assisted drop tower for Lab testing seismometer and examples of deceleration pulses.

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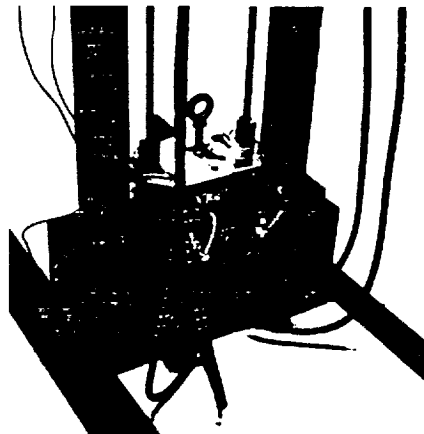
Addition of weight to
drop table to simulate
future seismometer



Release mechanism



Signal analyzer



Drop table and bungee
cords at point of impact

Figure 4 Photos of Bungee assisted drop tower.

We subjected the leaf spring sensor to impact impulses of 3 g secs (15,000 g at 0.2 ms, figure 3) for a variety of combinations of peak pulse and duration. It survived longitudinal shocks well but lateral shocks caused distortion of the frame supporting the springs. Components must be modified and the design changed until performance survival is guaranteed.

conclusion

Seismometers, many orders more sensitive than those on Viking, emplaced on Mars, will

detect marsquakes, meteorite impacts and, possibly, landslides. To identify the locations of events, and to correlate phases, at least 4 stations are required, 3 for location and a fourth for redundancy. To examine diverse geological sites, several different regions should be instrumented; A total of 12 sites with 2 stations per site would achieve these goals.

Emplacement by penetrator, with detachable forebody, achieves good coupling, isolation from surface temperature and wind pressure effects; but the high g loads risk the seismometer and probably rules out using an RTG.

Emplacement by hard-lander on the surface, could achieve fair coupling, if post-emplacement mechanisms are employed (such as driving in a spike or drilling). It will need special provision for isolation from temperature and wind pressure effects, which if only partially successful, will result in a short period narrow band station only. High g loads can be minimized, to less than several hundred g's, if a rough-lander is used.

Leaf spring, force-balance feedback, seismometers have the wide band-width, dynamic range, shock tolerance and sensitivity to be used in penetrators or surface landers. They are light but consume more power than narrow-band magnet-coil velocity transducers. We have tested a brass-board suspension design, which approaches the necessary ruggedness, but has about 1/3 the sensitivity of a state-of-the-art instrument.

Acknowledgements This work was supported by grant CS-04-89 from California Space Institute.

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6.2 SESSION B SUBMITTALS

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Session B, Submittal No. 1

Phil Knocke
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**A POLAR ORBIT FOR THE
MARS GLOBAL NETWORK MISSION**

PHIL KNOCKE

JPL

*A POLAR ORBIT FOR THE
MARS GLOBAL NETWORK MISSION*

Philip Knocke
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Presented to the Mars Global Network Mission Workshop
Jet Propulsion Laboratory, Pasadena Ca.
February 6-7, 1990

INTRODUCTION

The purpose of the Global Network Mission (GNM) is to deploy simple landers on the Martian surface in late 1998. The objective is to create a globally distributed network of ground stations which will collect environmental data, perhaps for as long as several years. The GNM presents unique mission design challenges, which are addressed by the following essay.

The GNM mission concept calls for two carrier spacecraft, each equipped with a number of simple landers. Some of the landers may be deployed from approach, either to reduce carrier mass prior to orbit insertion, or to reach latitudes not available from the carrier orbit. The remaining landers are deployed from orbit.

One configuration for the Global Network Mission was proposed in a report from the Exploration Precursors Task Team to the Office of Space Science and Applications.¹ This formed the basis of a previous orbit design for the GNM.² The following analysis uses this mission scenario as a point of reference, but results from the current study are generally applicable to a wide range of GNM mission variants.

FACTORS INFLUENCING MISSION DESIGN

The need to minimize the orbit insertion ΔV of the carrier implies that the carrier orbit be as elliptical as possible, and have a low periapse altitude. Elliptical orbits also

lead to lower de-orbit ΔV 's than circular orbits.

A number of other requirements act in concert to lay severe constraints on the orbit design for this mission. Among them is the need to distribute the landing sites globally. The overall goals of the mission, as well as guidance from the Mars Science Working Group, indicate a need to emplace landers near the Martian poles. This calls for an orbit capable of reaching latitudes of at least $\pm 80^\circ$. Coupled with this requirement is the need for good lighting angles at impact, to support descent imaging. Ideally, the sun elevations at impact would never exceed 30° or fall below 15° . An acceptable range of solar elevations is 10° to 45° .¹ The lighting conditions, coupled with the requirement for extensive latitudinal dispersal, constitute the major orbit design drivers.

In most cases, the lander is restricted to a given range of entry flight path angles. This has particular significance in the case of landers deployed from approach. The circumstances of the interplanetary trajectory, in particular the declination of the arrival asymptote, produce a minor circle of impact points which satisfy the desired entry angle. This leads to severe restrictions on the maximum north and south latitudes available to an approach lander. For example, a high negative approach declination produces rather low maximum northern latitudes at the desired entry angle. The only way to achieve impact at the North Pole in this case is to enter at prohibitively steep entry angles.² In addition, approach-deployed landers must accept whatever lighting conditions are available at their impact latitude.

ASSUMPTIONS

The current analysis uses the nominal GNM mission plan described in Reference 2. This specifies a launch period from December 6, 1998 to December 26, 1998, and an arrival period from September 22, 1999 to October 9, 1999. Entry interface was defined at an altitude of 125 km, and the nominal entry flight path angle at this point was taken to be -20° . The impact point was determined by propagating the free space trajectory from entry interface to an altitude of 10 km. Impact was assumed to occur directly beneath this point. (Atmospheric deceleration was not specifically addressed. The effects of drag would change the impact point by only a very few degrees along-track.) As mentioned earlier, this was only a reference scenario. The results are applicable to a range of entry angles and mission options.

The nominal deployment scenario described in Reference 2 was retained for this study. Figure 1 illustrates the deployment technique, in which the lander's de-orbit ΔV is applied tangential to the carrier's motion, and parallel to the entry velocity vector. This assures zero angle of attack at entry. The advantage of this mode of deployment is that no attitude sensors or attitude adjustments are required after deployment. All orbit-deployed landers are deployed from a fixed point in the carrier's orbit, and always impact at a fixed true anomaly with respect to the carrier's periapse location. As the carrier periapse moves due to nodal and apsidal rotation, the impact point moves along the surface of the target planet. The orbit must be chosen such that the nodal and apsidal motions place the impact points at favorable lighting conditions. Note that the maximum latitude available from orbit is equal to the orbital inclination. Longitudinal placement is achieved by making very small changes in the orbital period, causing the ground track to "walk" in longitude.

PREVIOUS ORBIT DESIGN

The nominal orbit design described in Reference 2 involves one carrier in a 45° inclined orbit, and a second carrier in a complementary, 135° retrograde orbit. Both carriers are in $1/5$ sol site-synchronous orbits with periapse altitudes of 200 km. Figure 2 shows a plot of sun elevation at impact vs. latitude of impact for the 45° orbit. As shown, immediately after insertion, the carrier can deploy landers at favorable sun elevation angles. In this orbit, there is a single sweep of deployment opportunities from 45°N to 45°S . The retrograde, 135° orbiter must wait between 70 and 150 days after arrival before deploying its landers. The retrograde orbiter sweeps once from 45°S to 45°N .

The advantage of the nominal orbit design is that some landers may be deployed immediately after arrival. This orbit does not allow easy attainment of high latitudes, however. In order to reach the North Pole, a lander would have to be deployed on approach, and enter the atmosphere at very steep entry angles (-43.9° to -49.8°).² A lander placed at the North Pole would also enter in darkness. Another factor to consider is the lack of deployment redundancy; there is only one deployment sweep from 45°N to 45°S . Favorable lighting angles do not occur again for several hundred days, and only for a narrow range of latitudes.

POLAR ORBIT

Figure 3 shows a plot of solar elevation at impact vs. latitude of impact for a carrier in a 1/5 sol orbit, with an inclination of exactly 90° and a periapse altitude of 275 km. The graph applies to a direct, periapse insertion from a northern approach at the start of the arrival period. Initially, the impact point is at the North Pole, which is in darkness. After waiting approximately 160 days, however, the impact point has moved to the Southern Hemisphere, and the lighting angles have moved into the acceptable range. Shortly thereafter, the impact point sweeps from the South Pole to the North Pole, remaining at good lighting angles. After the North Pole is reached, the impact points move south again, staying at reasonable lighting conditions until a latitude of 55°S is attained.

This situation occurs, in part, because the impact point moves from the South Pole to the North Pole as the Sun is moving from the Southern Hemisphere to the Northern Hemisphere. Figures 4 and 5 illustrate how the impact point follows the Sun. In addition, it is necessary that the orbit plane be placed properly with respect to the Sun, and that the rate of periapse advance be chosen to complement both the nodal movement with respect to the Sun, and the rate of change in solar declination. The 1/5 sol orbit is the most elliptical site-synchronous orbit with the required characteristics, and the 275 km periapse altitude provides the best lighting conditions for both the south-north sweep and the sweep from the North Pole to 55°S . The situation is similar at the end of the arrival period, although a small periapse rotation at insertion is required.

The advantages of such an orbit are evident. It allows landers to be placed anywhere on the Martian surface at reasonable lighting conditions and at the desired entry angle. A measure of redundancy is afforded by the second sweep from 90°N to 55°S . (This sweep could be used as backup in the event of failed landings on the first sweep.) The polar landers would be deployed from orbit instead of approach, and would enter at the nominal entry angle. The option exists to deploy all the landers from orbit, thereby eliminating the need for two deployment techniques, and avoiding the larger landing dispersion of approach-deployed landers.

The major disadvantage of this orbit design is the 160 day wait time required before lander deployment. This interval is largely unavoidable, as the orbit only slowly drifts into the required solar geometry. It should be noted, however, that for the 1998 opportunity, the wait interval allows the dust storm season to pass before first

deployment. The time could be used for other purposes as well, such as aeronomy measurements. The carrier could be placed in an orbit with a lower periapse, and then elevated to the 275 km altitude for a small investment in ΔV .

CONCLUSIONS

A 1/5 sol, polar orbit with a periapse altitude of 275 km offers the best circumstances for orbital deployment of the Global Network Mission landers. It allows easy polar access at nominal entry angles, and global dispersal of landing sites at lighting angles suitable for descent imaging. The polar orbit allows the option of deploying all the landers from orbit. A wait interval of 160 days after arrival is required before deployment can commence.

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2. Knocke, P., "Global Network Mission Analysis", NASA Jet Propulsion Laboratory Interoffice Memorandum 312/89.2-1581, December 15, 1989.

LANDER DEPLOYMENT FROM ORBIT

De-orbit ΔV applied tangentially and parallel to entry velocity vector.

Entry Interface: 125 km altitude, -20° entry flight path angle

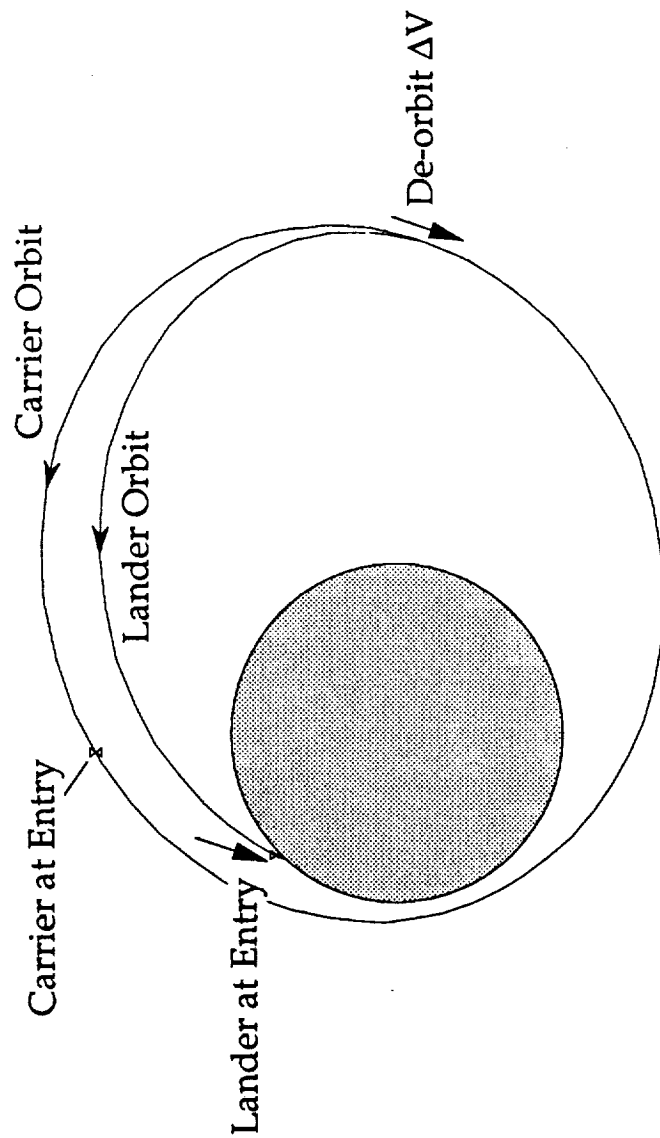


Figure 1

45° ORBIT: SUN ELEVATION VS. IMPACT LATITUDE

i=45°, 1/5 sol, 200 km periapse altitude

North Approach, Start of Arrival Period

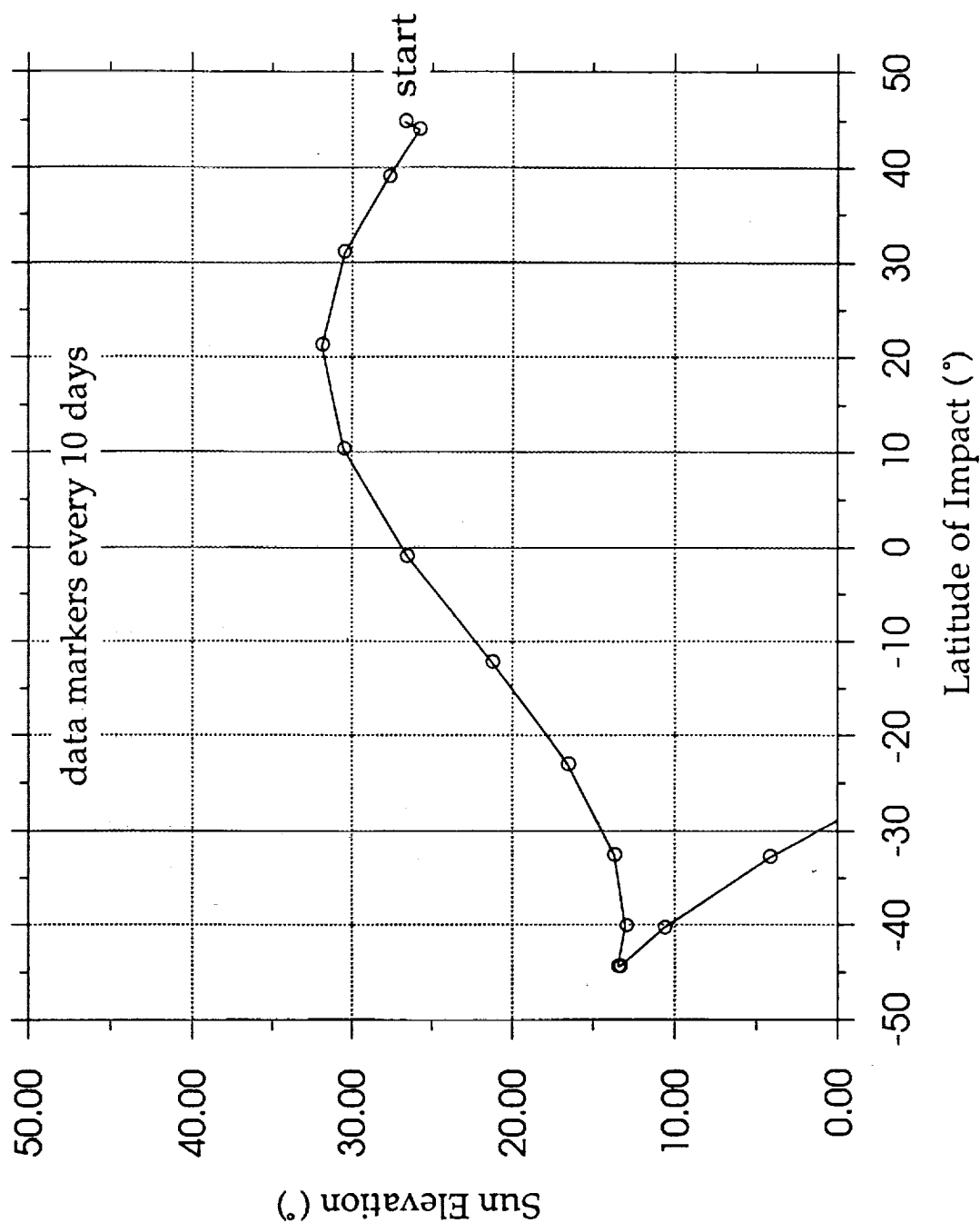
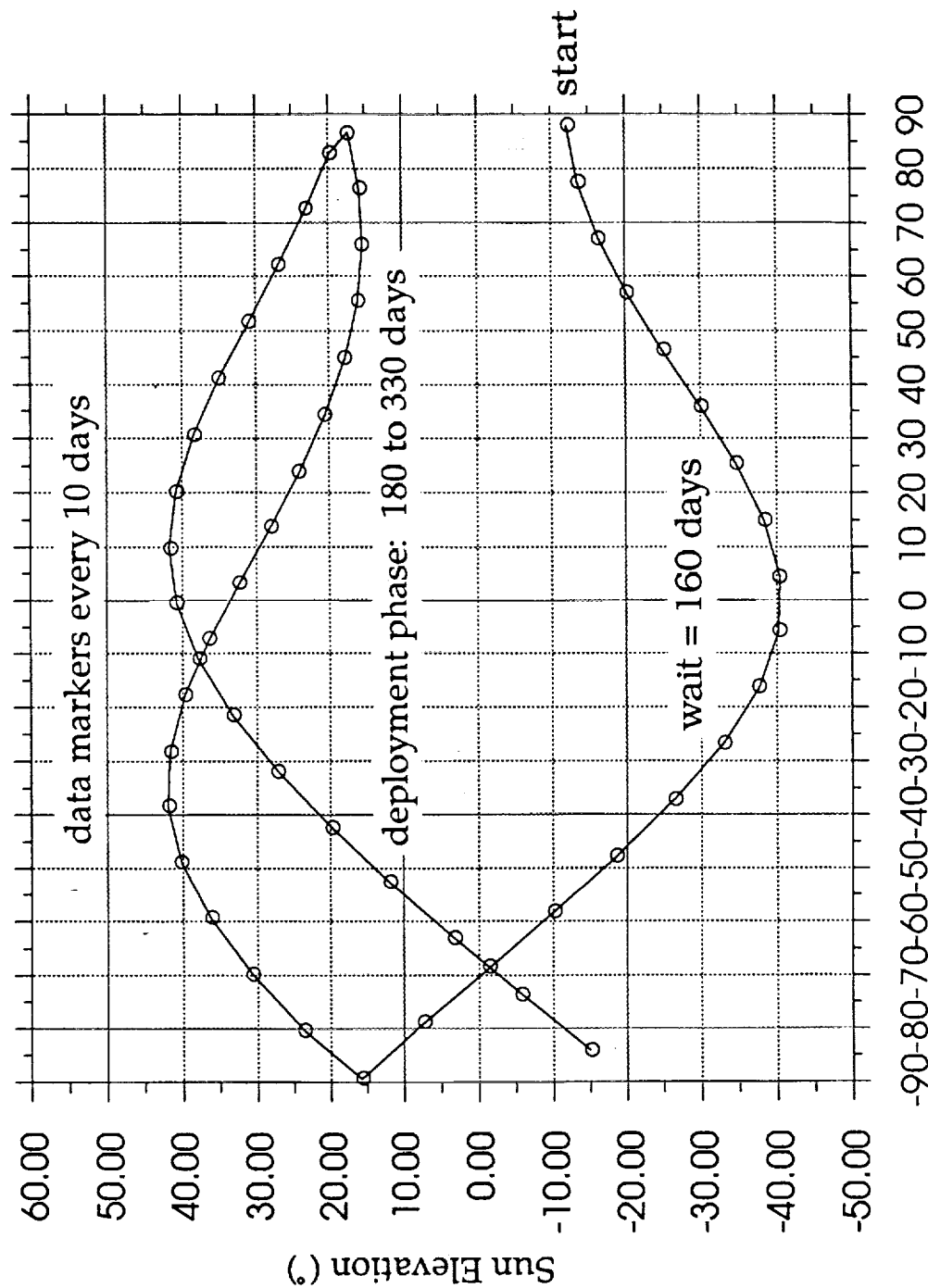


Figure 2

POLAR ORBIT: SUN ELEVATION VS. IMPACT LATITUDE

$i=90^\circ$, 1/5 sol, 275 km periapse altitude

North Approach, Start of Arrival Period



Latitude of Impact (°)

Figure 3

POLAR ORBIT: 200 DAYS AFTER ARRIVAL (2000/04/14)

ENTRY: 125 km, -20°, 4.2 km/s

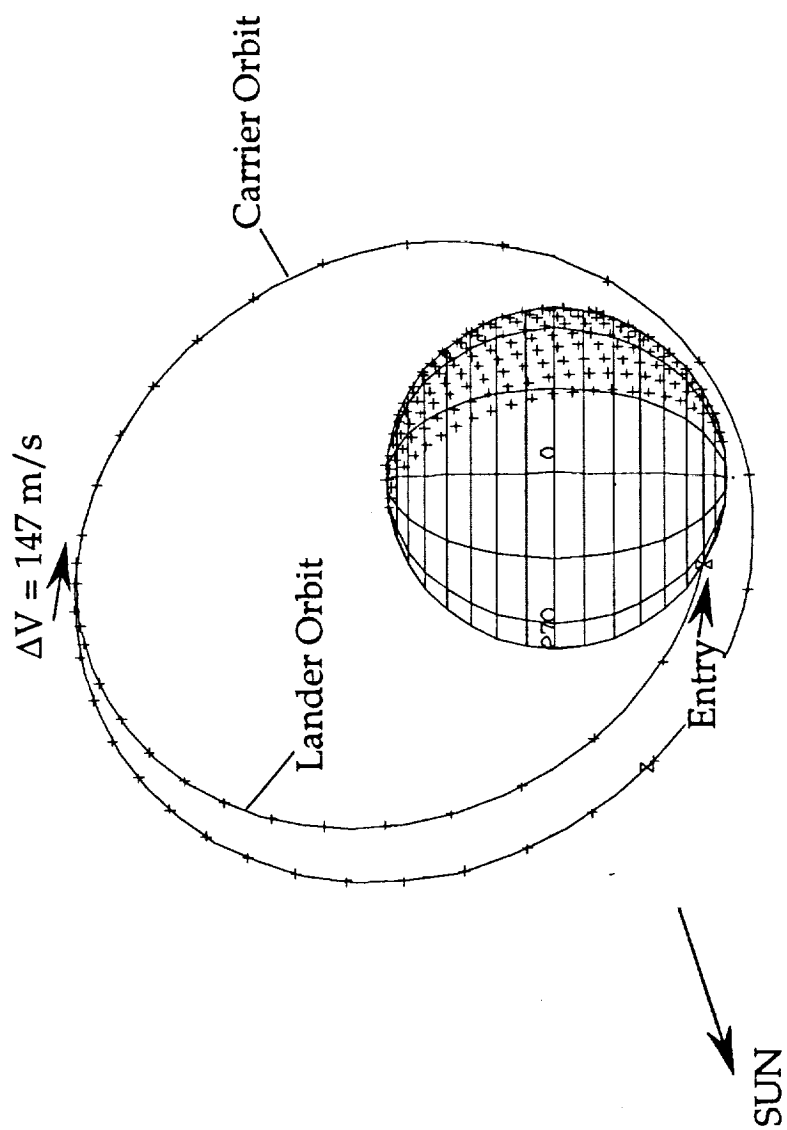


Figure 4

**POLAR ORBIT: 300 DAYS AFTER ARRIVAL
(2000/07/25)**

ENTRY: 125 km , - 20° , 4.2 km/s

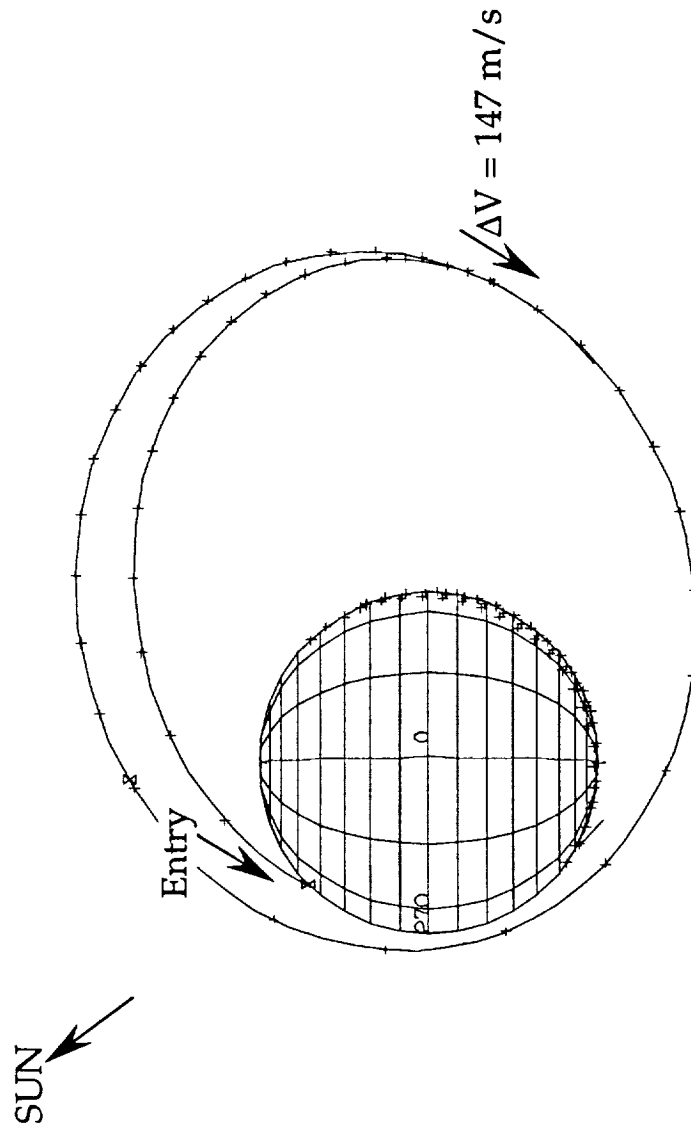


Figure 5

Session B, Submittal No. 2

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MARS GLOBAL NETWORK SURFACE LANDERS
SYSTEMS DESIGN OPTIONS

ARDEN ALBEE

CALTECH

JIM BURKE
ROBERT MOSTERT

JPL

SYSTEM CONSIDERATIONS FOR MARS NETWORK SURFACE LANDERS

OBJECTIVES:

- PRIORITIZE MISSION GOALS RELATIVE TO 90 DAY STUDY (REF: 12/15/89)
- IDENTIFY MISSION CONSTRAINT(S)
 - (a) Mission requires multiple landers for several sites ((O) = 20)
 - (b) Mission requires long lifetime for successful data acquisition
- NEED TO CONSIDER MEANS TO:
 - (a) Increase number of lander sites (90 day study = 12)
 - (b) Reach higher latitudes (90 day study constrained by $\alpha = 15-20$ deg)
 - (c) Take advantage of prioritized goals to assess system designs

APPROACH:

- ASSESS IMPACT OF EACH POSSIBLE GOAL ON SYSTEM DESIGN
- SKETCH SYSTEM CONCEPTS COMPATIBLE WITH DIFFERENT SETS OF GOALS: ALTERNATIVE OR DUAL DESIGNS
 - (a) Raises question of doing everything with one design
 - (b) Two simple designs vs. one complex design
- DEFINE DEVELOPMENTS NEEDED FOR SYSTEM OPTIONS CONSIDERED PROMISING

DESIRED RESULT:

- BETTER KNOWLEDGE OF GOALS \longleftrightarrow DESIGN POSSIBILITIES;
- \therefore BETTER KNOWLEDGE OF DEVELOPMENT PRIORITIES

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GOAL OBJECTIVES, SETS AND CONSTRAINTS

OBJECTIVES GROUP INTO THREE SETS:

CONSTRAINT(S)

A) SIMPLEST SURFACE LANDER:

LONG-LIFE METEOROLOGY
LONG-LIFE SEISMOLOGY

RTGs vs. solar panels; small data rate, large quantity; power and thermal considerations

SURFACE CHEMISTRY

Additional feature to simplest lander

B) ADDITION TO SIMPLE LANDER (SUBSET OF (A)):

DESCENT IMAGING
POST-LANDING IMAGING

Lighting conditions; high data rate or store/readout; modest total quantity; relies on impact survival

C) OTHER ADDITIONS TO SIMPLE LANDER (W/WO IMAGING):

SUBSURFACE SEISMOLOGY
SUBSURFACE CHEMISTRY
SUBSURFACE VOLATILES

Small data rate and quantity; long-life goal; power and thermal considerations; requires sample or instrument to be subsurface; favors penetrators

SOME GOAL IMPLICATIONS

A) POLAR LANDINGS

High entry angles; more TPS and aerodeceleration demand.

B) MAXIMUM DISPERSAL AND NUMBER OF SITES

Favors singly launched landers from S/C; favors smaller, simpler landers.

C) CONSTRAINTS ON EXPERIMENT DESIGN

Size, mass, volume, power, thermal, lifetime, data compression.

GOALS GROUP INTO THREE CLASSES

(1) WANT TO BE BELOW SURFACE (M's), CAN BE SHORT-LIVED, SMALL DATA RATE AND QUANTITY

(EXAMPLES: CHEMISTRY, VOLATILES DETECTION)

(2) MUST BE LONG-LIVED (~YRS), CAN BE AT SURFACE, SMALL DATA RATE, LARGE TOTAL QUANTITY

(EXAMPLES: SEISMOLOGY, METEOROLOGY)

(3) MORE COMPLEX, CAN BE SHORT-LIVED, EITHER HIGH DATA RATE OR STORE/READOUT, MODEST TOTAL QUANTITY

(EXAMPLES: DESCENT IMAGING, POST-LANDING IMAGES)

SYSTEM OPTIONS

- VIKING/SURVEYOR-TYPE LARGE SOFT LANDERS - RULED OUT BASED ON SIZE AND BY NUMBER DESIRED FOR MISSION (NETWORK DESIGN)
- SOVIET-TYPE "ROUGH" LANDERS - TOUCHDOWN PROXIMITY RETRO
- CSAD-TYPE "SLOW IMPACTERS" (E.G. RANGER 3-5)
- DESCENT IMAGERS THAT CRASH (E.G. RANGER 6-9)
- DESCENT IMAGERS DESIGNED TO SURVIVE AND SEND IMAGE(S) AFTER LANDING
- RUGGED LANDERS (HIGH "G") = PENETRATORS

⇒ NETWORK MISSION CONCEPT REQUIRES MANY SITES

POST-LANDING SURVIVAL (IT CAN BE BRIEF)
DESIRABLE AT THE NETWORK SITES

∴ CHOICE IS NARROWED TO "HIGH G (PENETRATOR AFTERBODY)" VS
"MODERATE G" LANDERS WHERE "HI" $> 10^3$; "MODERATE" 10's OF G's

PROS AND CONS OF ALTERNATIVE LANDER TECHNIQUES

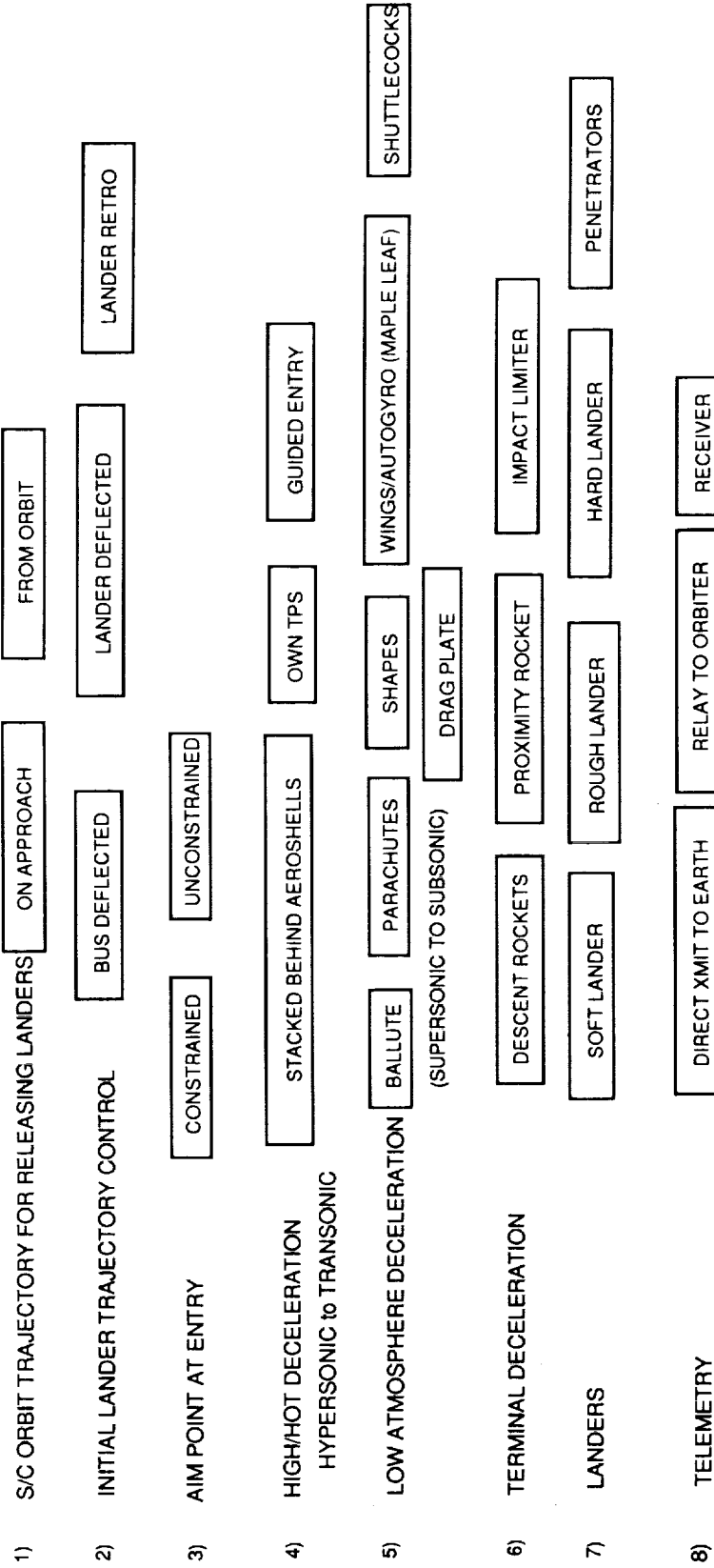
	<u>PROS</u>	<u>CONS</u>
o BUS DEFLECTION	GOOD TARGETING; SIMPLIFIES LANDERS; USES EXISTING BUS SUBSYSTEMS	QUARANTINE
o LANDER DEFLECTION	QUARANTINE; SIMPLIFIES SEQ.	LANDERS MAY NOT ALL BE THE SAME
o NARROW ENTRY ANGLE TOLERANCE	SIMPLIFIES TPS DESIGN	MAY NOT REACH POLES
o WIDE ANGLE TOLERANCE	EXPANDS REACHABLE LATITUDES	MORE TPS, AERODECEL.
o STACKED ON AEROSHELL	SIMPLIFIES LANDERS	DISPERSAL IN PAIRS, SETS
o TPS FOR EACH LANDER	MORE SITES REACHABLE	COMPLICATES LANDERS
o LIFTING ENTRY	MORE DISPERSION	LARGER ERR. ELLIPSES
o PARACHUTES	SIMPLE	MAY LIMIT ANGLE TOL.
o BALLUTES	MORE ANGLE TOLERANCE	DIFFICULT
o AUTOGYROS	MAY ALLOW GOOD IMAGING	NEEDS R & D
o AEROSHELL/(L/D) SHAPES	COMBINES FUNCTIONS	MAY AFFECT PACKING
o IMPACT LIMITER	WELL TESTED	WITHSTANDS 100'S G's
o ROUGH OR HARD LANDERS	SIMPLEST	SOME MAY FAIL
o PENETRATORS	PACKING	LONG-LIFE POWER, THERMAL
o TERMINAL ROCKETS	WELL USED	ADDS MASS TO SYSTEM

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A SUMMARY OF GOAL IMPLICATIONS

o DESCENT IMAGING	HIGH-RATE TELEMETRY AND/OR STORAGE, BUT STORAGE RELIES ON IMPACT SURVIVAL
o POST-LANDING IMAGING	RELIES ON IMPACT SURVIVAL
o LONG LIFE	PRIMARY IMPLICATION IS ON LANDER POWER/THERMAL SYSTEMS
o POLAR LANDINGS	HIGH ENTRY ANGLES; THUS MORE TPS AND AERODECELERATION DEMAND (NOTE: ENTRY "G's" MAY STILL BE LESS THAN TERMINAL DECELERATION "G's")
o MAXIMUM DISPERSAL	FAVORS NOT ENTERING IN PAIRS OR SETS AS A RESULT OF BEING STACKED ON AEROSHELLS
o MAXIMUM NUMBER OF TOTAL SITES DESIRABLE	FAVORS SMALLER, SIMPLER LANDERS
o SUBSURFACE MEASUREMENTS	FAVORS PENETRATORS, EITHER DROPPING IN OR DRIVEN IN
o CONSTRAINTS ON EXPERIMENTS	SIZE, MASS, VOLUME, POWER, THERMAL, LIFETIME, DATA COMPRESSION

SYSTEMS DESIGN TREE



MISSION GOALS:

SURFACE OBSERVATIONS

- 1) LONG LIFE SEISMOLOGY
- 2) LONG LIFE METEOROLOGY
- 3) POST-LANDING IMAGING
- 4) SURFACE CHEMISTRY

SUBSURFACE OBSERVATIONS

- 1) SHORT LIFE SEISMOLOGY
- 2) SHORT LIFE METEOROLOGY
- 3) SUBSURFACE CHEMISTRY
- 4) SUBSURFACE VOLATILES

90 DAY STUDY RESULTS

1)	S/C ORBIT TRAJECTORY FOR RELEASING LANDERS	ON APPROACH	FROM ORBIT	
2)	INITIAL LANDER TRAJECTORY CONTROL	BUS DEFLECTED	LANDER DEFLECTED	LANDER RETRO
3)	AIM POINT AT ENTRY	CONSTRAINED	UNCONSTRAINED	
4)	HIGH/HOT DECELERATION HYPERSONIC to TRANSONIC	STACKED BEHIND AEROSHHELLS	OWN TPS	GUIDED ENTRY
5)	LOW ATMOSPHERE DECELERATION	BALLUTE (SUPERSONIC TO SUBSONIC)	PARACHUTES SHAPES	WINGS/AUTOGYRO (MAPLE LEAF) SHUTTLECOCKS
6)	TERMINAL DECELERATION	DESCENT ROCKETS	PROXIMITY ROCKET	IMPACT LIMITER
7)	LANDERS	SOFT LANDERS	ROUGH LANDERS	HARD LANDERS PENETRATORS
8)	TELEMETRY	DIRECT XMIT TO EARTH	RELAY TO ORBITER	RECEIVER
ISSUE				
BUS DEFLECTION		PRO	VS.	CON
LANDER DEFLECTION		GOOD TARGETING, SIMPLIFIES DESIGNS, USE OF EXISTING BUS SYSTEM QUARANTINE, SIMPLIFIES SEQUENCE		QUARANTINE REQUIRES DELTA V AND SPIN CAPABILITY
NARROW ENTRY ANGLE		SIMPLIFIES AEROSHELL TPS DESIGN		DIFFICULT TO REACH POLES
STACKED ON AEROSHHELLS		SIMPLIFIES LANDER		DISPERSAL IN PAIRS OR SETS
PARACHUTES		SIMPLE		LIMITS ANGLE TOLERANCE
PENETRATOR		SIMPLIFIES SUBSURFACE OBSERVATIONS		DIFFICULT; POWER AND THERMAL CONSIDERATIONS

OVERALL DISADVANTAGE: MISSION HAS ONLY 6 PAIRS OF LANDERS; EACH PAIR TO BE DELIVERED TO THE SAME REGION

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SAIC 1970'S RESULTS

- 1) SC ORBIT TRAJECTORY FOR RELEASING LANDERS

ON APPROACH

FROM ORBIT
- 2) INITIAL LANDER TRAJECTORY CONTROL

BUS DEFLECTED

LANDER DEFLECTED

LANDER RETRO
- 3) AIM POINT AT ENTRY

CONSTRAINED

UNCONSTRAINED
- 4) HIGH/HOT DECELERATION
HYPERSONIC to TRANSONIC

STACKED BEHIND AEROSHHELLS

OWN TPS

GUIDED ENTRY
- 5) LOW ATMOSPHERE DECELERATION

PARACHUTES

BALLUTE

SHAPES

WINGS/AUTOGYRO (MAPLE LEAF)

SHUTTLECOCKS

(SUPERSONIC TO SUBSONIC)
- 6) TERMINAL DECELERATION

DESCENT ROCKETS

PROXIMITY ROCKET

IMPACT LIMITER
- 7) LANDERS

SOFT LANDERS

ROUGH LANDERS

HARD LANDERS

PENETRATORS
- 8) TELEMETRY

DIRECT XMIT TO EARTH

RELAY TO ORBITER

RECEIVER

ISSUE	PRO	VS.	CON
LANDER DEFLECTION	QUARANTINE, SIMPLIFIES SEQUENCE		REQUIRES DELTA V
NARROW ENTRY ANGLE	SIMPLIFIES TPS DESIGN		DIFFICULT TO REACH POLES
TPS FOR EACH LANDER	RANGE OF SITES REACHABLE AVAILABLE		COMPLICATES LANDERS
BALLUTES	MORE ANGLE TOLERANCE		DIFFICULT
PENETRATORS	SIMPLIFIES SUBSURFACE OBSERVATIONS		DIFFICULT; SHORT-LIVED

OVERALL DISADVANTAGE: MISSION DESIGNED TO HAVE ONLY 4-6 LANDERS; SHORT LIFETIME

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AMES 1977 RESULTS

- 1) S/C ORBIT TRAJECTORY FOR RELEASING LANDERS

ON APPROACH

FROM ORBIT
- 2) INITIAL LANDER TRAJECTORY CONTROL

BUS DEFLECTED

LANDER DEFLECTED

LANDER RETRO
- 3) AIM POINT AT ENTRY

CONSTRAINED

UNCONSTRAINED
- 4) HIGH/HOT DECELERATION
HYPERSONIC to TRANSONIC

STACKED BEHIND AEROSHELLS

OWN TPS

GUIDED ENTRY
- 5) LOW ATMOSPHERE DECELERATION

BALLUTE

PARACHUTES

SHAPES

WINGS/AUTOGYRO (MAPLE LEAF)

SHUTTLECOCKS

(SUPERSONIC TO SUBSONIC)

DRAG PLATE
- 6) TERMINAL DECELERATION

DESCENT ROCKETS

PROXIMITY ROCKET

IMPACT LIMITER
- 7) LANDERS

SOFT LANDERS

ROUGH LANDERS

HARD LANDERS

PENETRATORS
- 8) TELEMETRY

DIRECT XMIT TO EARTH

RELAY TO ORBITER

RECEIVER

ISSUE	PRO	VS.	CON
LANDER DEFLECTION	QUARANTINE, SIMPLIFIES SEQUENCE		RETRO FROM SPACECRAFT; REQUIRES OWN TPS
NARROW ENTRY ANGLE	SIMPLIFIES TPS DESIGN		DIFFICULT TO REACH POLES
DRAG PLATE	ALLOWS ONE LANDER PER PLATE		DIFFICULT
PENETRATORS	SIMPLIFIES SUBSURFACE OBSERVATIONS		DIFFICULT; POWER AND THERMAL CONSIDERATIONS

OVERALL DISADVANTAGE: MISSION DESIGNED TO HAVE ONLY 3 LANDERS; DIFFICULT TO ESTABLISH A TRUE GLOBAL NETWORK

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VIKING RESULTS

1)	S/C ORBIT TRAJECTORY FOR RELEASING LANDERS	ON APPROACH	FROM ORBIT	
2)	INITIAL LANDER TRAJECTORY CONTROL	BUS DEFLECTED	LANDER DEFLECTED	LANDER RETRO
3)	AIM POINT AT ENTRY	CONSTRAINED	UNCONSTRAINED	
4)	HIGH/HOT DECELERATION HYPERSONIC to TRANSONIC	STACKED BEHIND AEROSHHELLS	OWN TPS	GUIDED ENTRY
5)	LOW ATMOSPHERE DECELERATION	BALLUTE	PARACHUTES	SHAPES
		(SUPERSONIC TO SUBSONIC)		DRAG PLATE
6)	TERMINAL DECELERATION	DESCENT ROCKETS	PROXIMITY ROCKETS	
7)	LANDERS	SOFT LANDERS	ROUGH LANDERS	HARD LANDERS
				PENETRATORS
8)	TELEMETRY	DIRECT XMIT TO EARTH	RELAY TO ORBITER	RECEIVER
	ISSUE	PRO	VS.	CON
	BUS DEFLECTION	GOOD TARGETING, SIMPLIFIES LANDER		QUARANTINE
	WIDE ANGLE TOLERANCE	EXPANDS REACHABLE LATITUDES		MORE TPS NECESSARY
	AEROSHELL	ONE LANDER PER AEROSHELL, SIMPLIFIES ATMOSPHERIC TPS		AFFECTS PACKAGING
	SOFT LANDER	HAS BEEN ACCOMPLISHED; LONG-LIVED; SIMPLIFIES POWER CONSIDERATIONS		LARGE; DIFFICULT TO GET DESIRED NUMBER

OVERALL DISADVANTAGE: MISSION AS DESIGNED HAD ONLY 1 LANDER PER S/C; DOES NOT CONSTITUTE A GLOBAL NETWORK W/O SEVERAL SPACECRAFT

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MARS 3 RESULTS

1)	S/C ORBIT TRAJECTORY FOR RELEASING LANDERS	ON APPROACH	FROM ORBIT
2)	INITIAL LANDER TRAJECTORY CONTROL	BUS DEFLECTED	LANDER DEFLECTED LANDER RETRO
3)	AIM POINT AT ENTRY	CONSTRAINED	UNCONSTRAINED
4)	HIGH/HOT DECELERATION HYPERSONIC to TRANSONIC	STACKED BEHIND AEROSHELLS	OWN TPS GUIDED ENTRY
5)	LOW ATMOSPHERE DECELERATION	BALLUTE (SUPERSONIC TO SUBSONIC)	PARACHUTES SHAPES WINGS/AUTOGYRO (MAPLE LEAF) SHUTTLECOCKS
6)	TERMINAL DECELERATION	DESCENT ROCKETS	PROXIMITY ROCKET IMPACT LIMITER
7)	LANDERS	SOFT LANDERS	ROUGH LANDERS HARD LANDERS PENETRATORS
8)	TELEMETRY	DIRECT XMIT TO EARTH	RELAY TO ORBITER RECEIVER
ISSUE			
	LANDER DEFLECTION	PRO	VS. CON
	NARROW ENTRY ANGLE	QUARANTINE, SIMPLIFIES SEQUENCE	REQUIRES DELTA V AND SPIN CAPABILITY
	BALLUTE	SIMPLIFIES TPS DESIGN	MAY BE DIFFICULT TO REACH POLES
	PARACHUTES	MORE ANGLE TOLERANCE	DIFFICULT
	TPS FOR EACH LANDER	SIMPLE	LIMITS ANGLE TOLERANCE
		RANGE OF SITES REACHABLE	COMPLICATES LANDERS

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Session B, Submittal No. 3

Alan L. Friedlander
Science Applications International Corporation

**A GLOBAL VIEW OF LANDER-TO-ORBITER
COMMUNICATIONS ACCESSIBILITY
FOR A MARS GLOBAL NETWORK MISSION**

**ALAN FRIEDLANDER
SAIC**

A Global View of Lander-to-Orbiter Communications Accessibility for a Mars Global Network Mission

Alan Friedlander
Science Applications International Corporation

Presented to Mars Global Network Mission Workshop
Jet Propulsion Laboratory, February 6-7, 1990

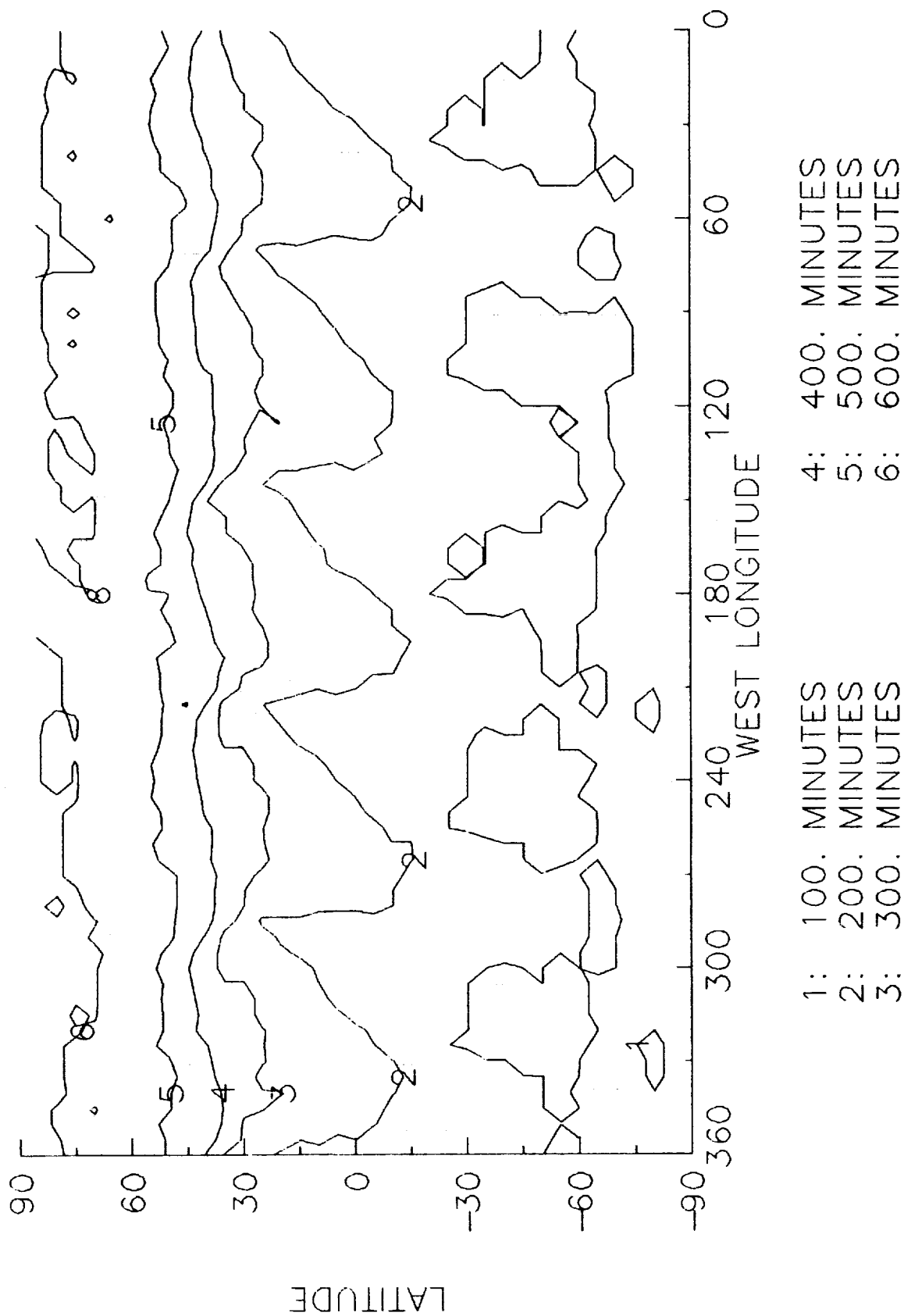
Given the mission objective to deploy a number of small landers to the surface of Mars at various latitude/longitude locations, it is of interest to obtain a global perspective of the communications link geometry between the landers and a data relay orbiter. Specifically, the question to be answered is what is the total time interval over one martian day (1 sol) that a lander at any given latitude and longitude can communicate data to the orbiter. Results should be obtained for more than one elevation angle constraint (lander antenna design issue), and also for several time points into the mission since the orbiter's periapsis location moves under the influence of Mars oblateness perturbation. This paper presents such information in terms of global contour maps of available communications time per sol summed over all orbiter passes on that day. Global data of this type complements more detailed local site data such as communications range and elevation vs time per pass.

The reference mission launched in 1998 arrives at Mars in late September 1999 and the orbiter is placed into a polar orbit (90 deg inclination) with periapsis altitude of 275 km, apoapsis altitude of 6903 km, and orbit period of 1/5 sol. Periapsis latitude is initially at 27 deg N and moves southward at the rate of about 1 deg/sol. Landing sites for orbit deployment are displaced about 56 deg from the orbiter's periapsis, thus starting near the north pole and moving southward. If the landers have descent imaging capability with a requirement for low sun elevation angles between 15 deg and ~ 30 deg, then the first deployment from orbit must be delayed until sol # 180 after arrival. Thereafter, all landing site latitudes are accessible with good values of sun angle. Pole-to-pole coverage is accomplished in about 6 months.

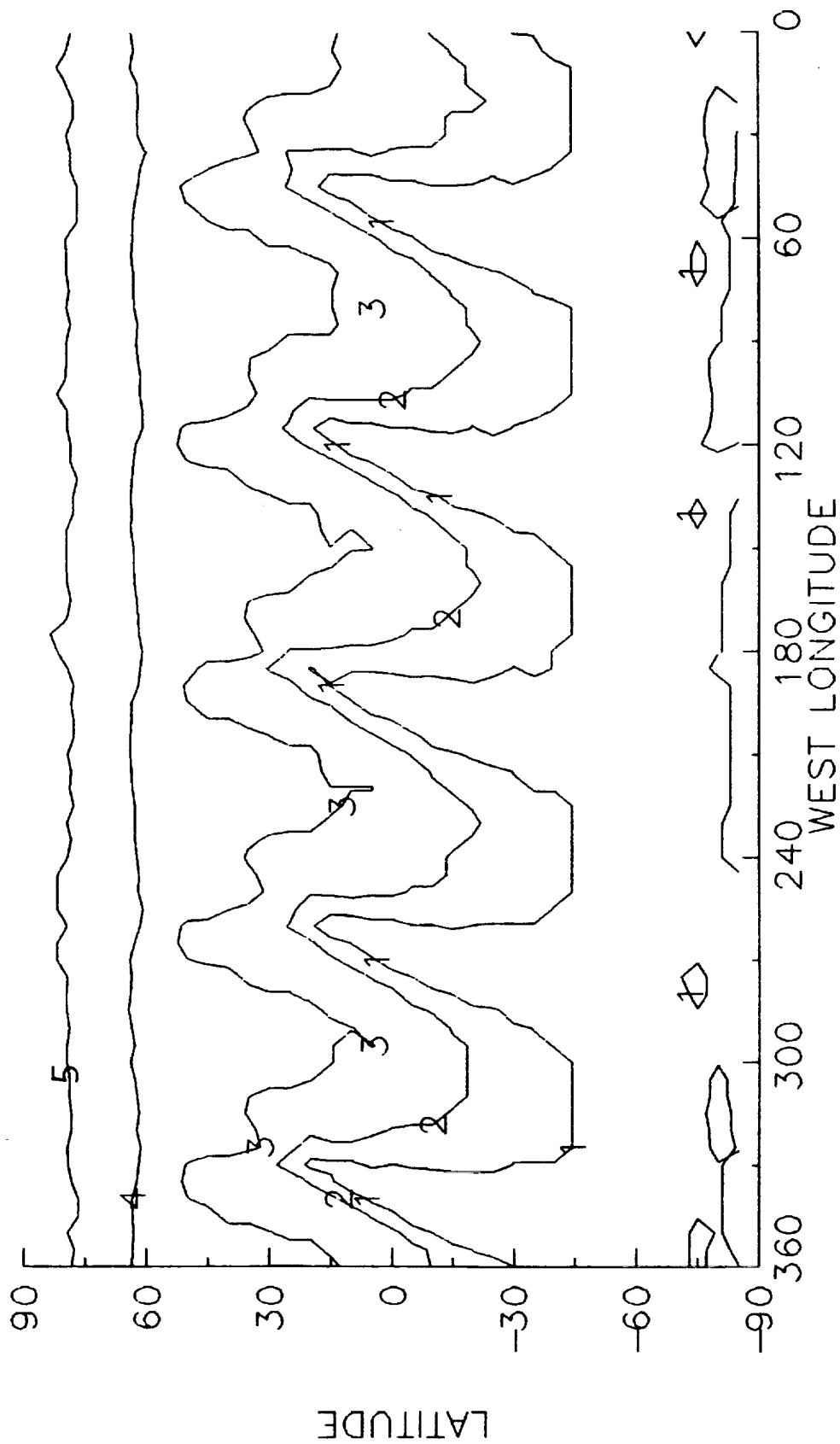
Communications time contour maps are included here for sol #'s 180, 232, 318, 361, and 404 corresponding to orbiter periapsis latitudes of 35 S, 90 S, equatorial, 45 N, and 90 N. For each of these days, there is a map for both a 15 deg and 45 deg minimum elevation constraint on the lander-to-orbiter line of sight. The jagged appearance of the contour lines is due to computational resolution in interpolating between a finite number of latitude/longitude grid points. Although the contours should really be smooth, the general information content is represented by the lower resolution maps shown here. An example of the tabulated, finite-grid data points is also given.

Communication with all sites is possible at the 15 deg elevation constraint, at times only for several minutes per sol but more generally for a much longer time up to 14 hours per sol. Significantly less time is available with a 45 deg elevation constraint, and at certain times in the mission some localized regions of the planet are inaccessible. Still, one may conclude that the reference orbit selection will support a more than adequate communications link through the mission timeline with landers emplaced at any desired location on Mars.

COMM AVAIL. PER SOL, SOL #180 (4/1/2000)
 HP=275KM,P=1/5 SOL,I=90DEG,15DEG ELV LIM
 (PERIAPSIS AT 35°S)

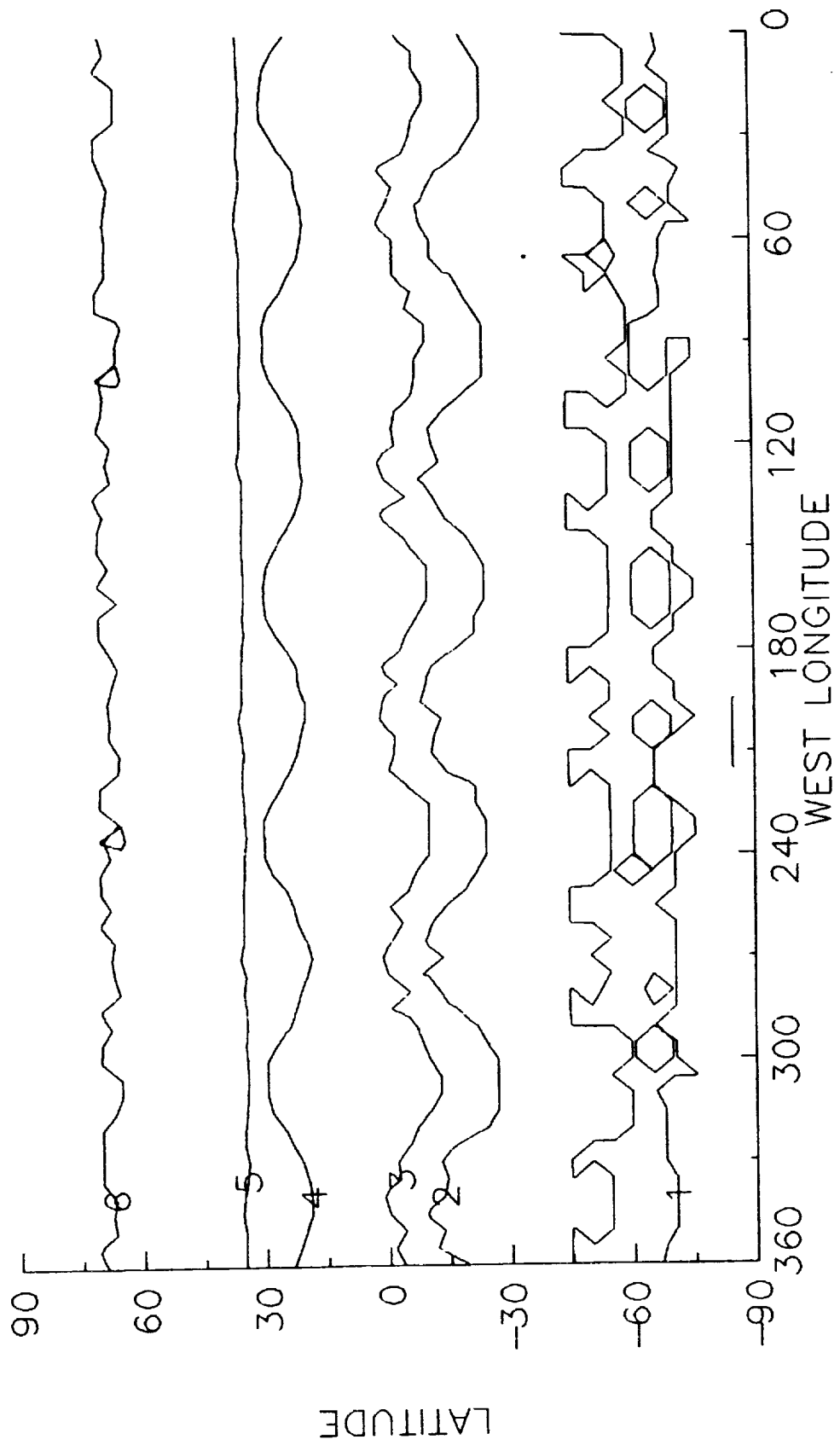


COMM AVAIL. PER SOL, SOL #180 (4/1/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 45DEG ELV LIM
 (PERIAPSIS AT 35°S)



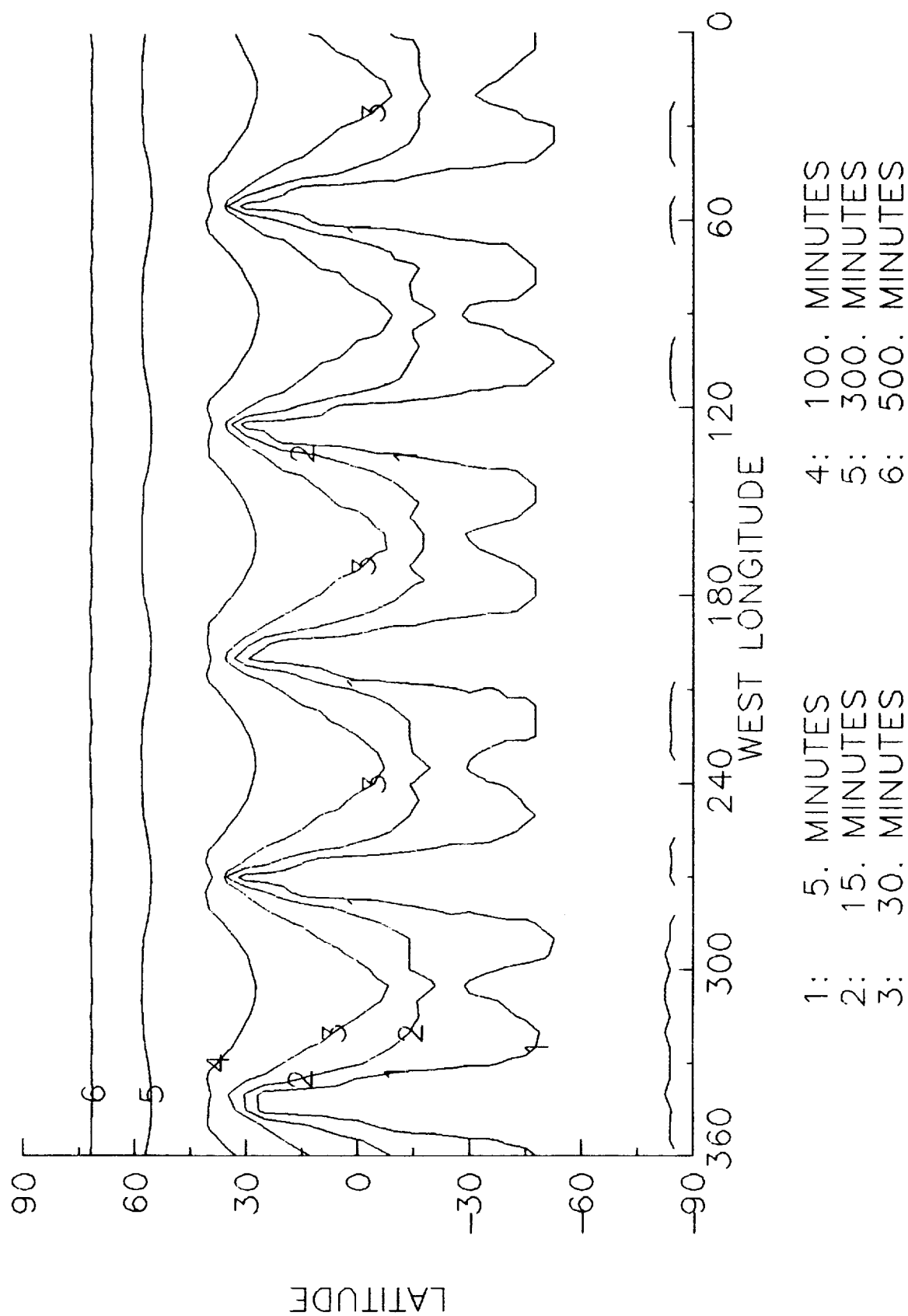
1: 30. MINUTES 4: 200. MINUTES
 2: 60. MINUTES 5: 300. MINUTES
 3: 100. MINUTES

COMM AVAIL. PER SOL,SOL #232 (5/24/2000)
 HP=275KM,P=1/5 SOL,I=90DEG,15DEG ELV LIM
 (PERIAPSIS AT 90°S)

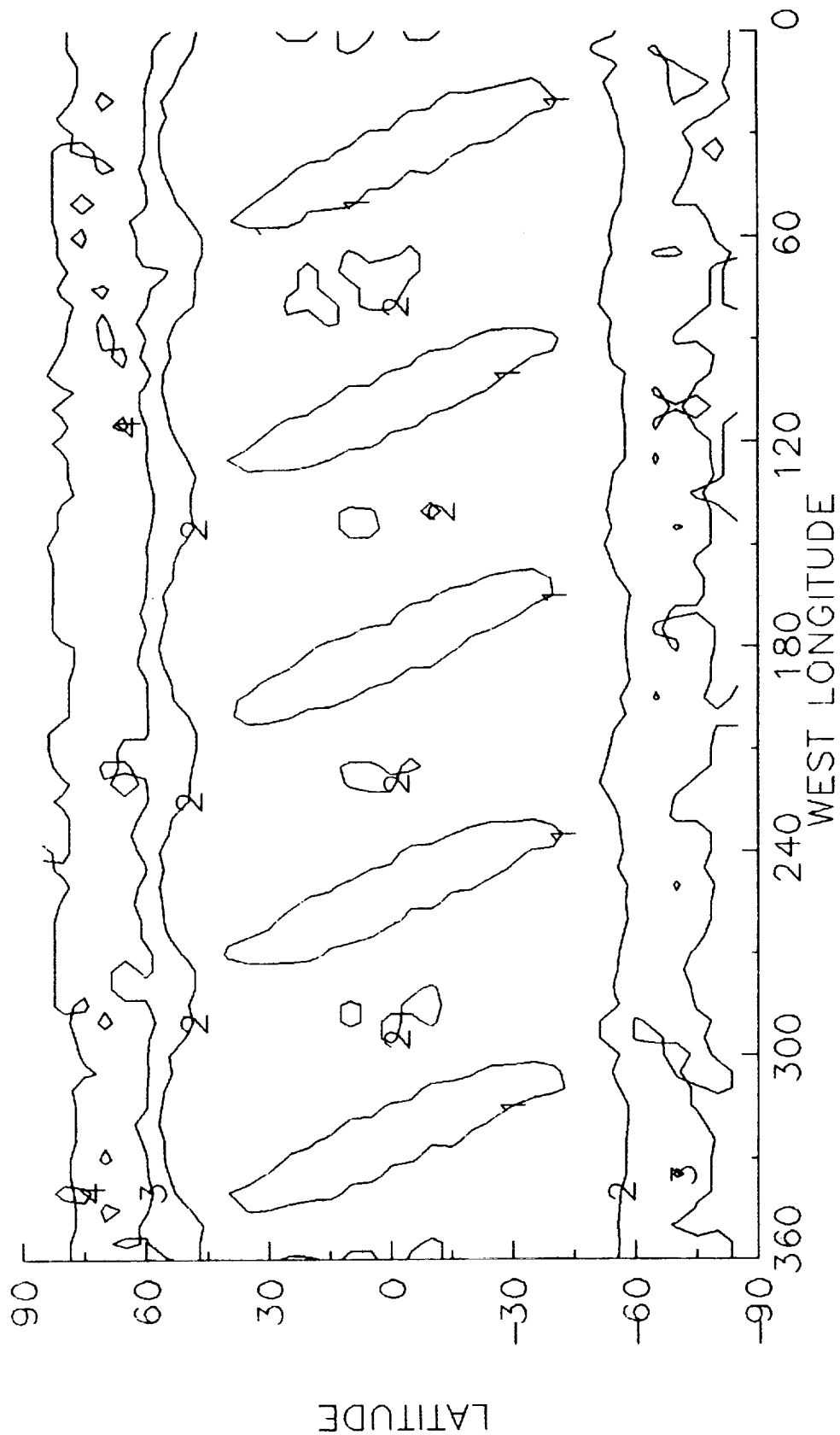


- 1: 15. MINUTES
- 2: 60. MINUTES
- 3: 100. MINUTES
- 4: 300. MINUTES
- 5: 500. MINUTES
- 6: 800. MINUTES

COMM AVAIL. PER SOL, SOL #232 (5/24/2000)
 HP=275KM, P=1/5 SOL, l=90DEG, 45DEG ELV LIM
 (PERIAPSIS AT 90°S)

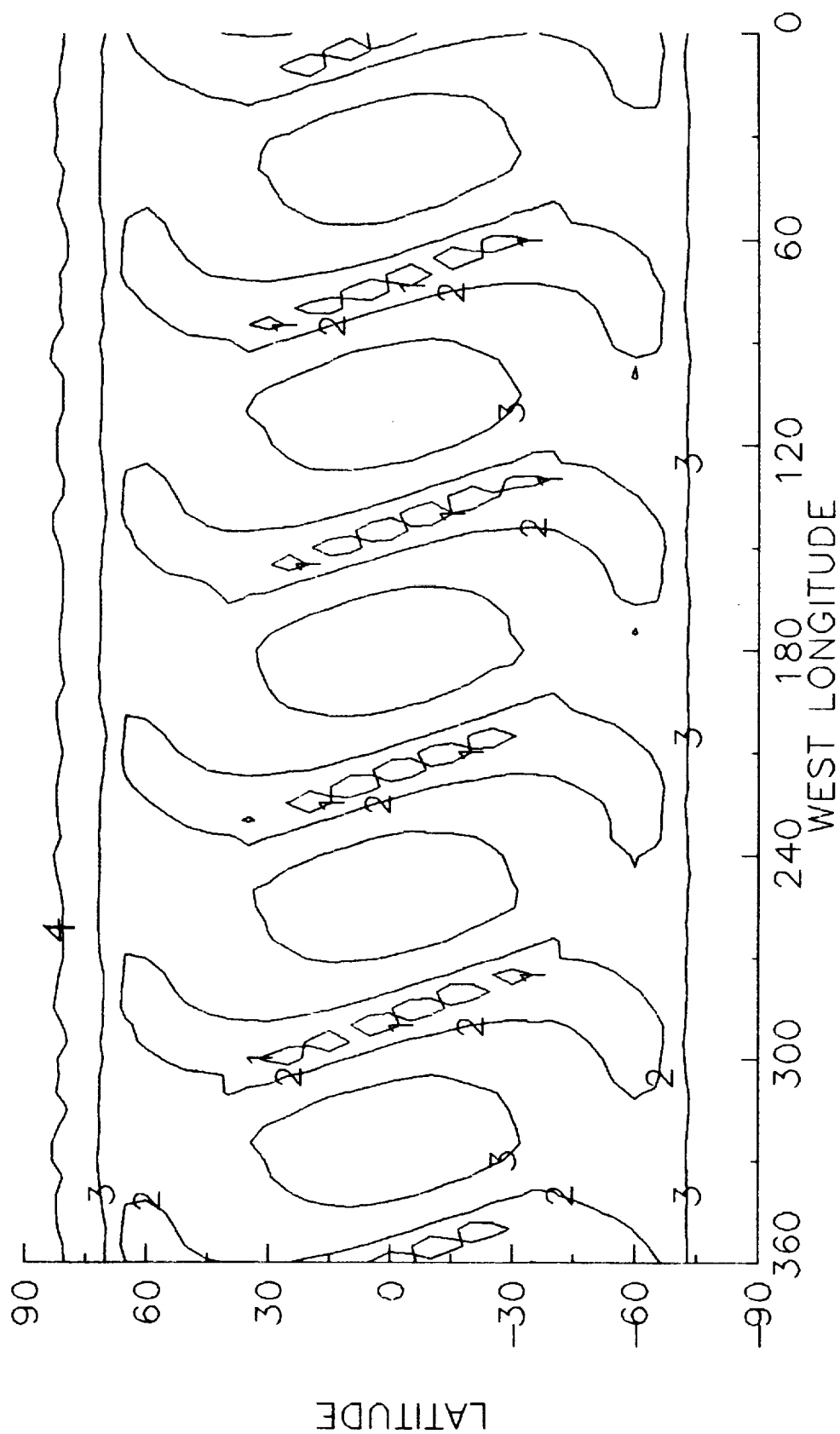


COMM AVAIL. PER SOL, SOL #318 (8/20/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 15DEG ELV LIM
 (PERIAPSIS AT EQUATOR)



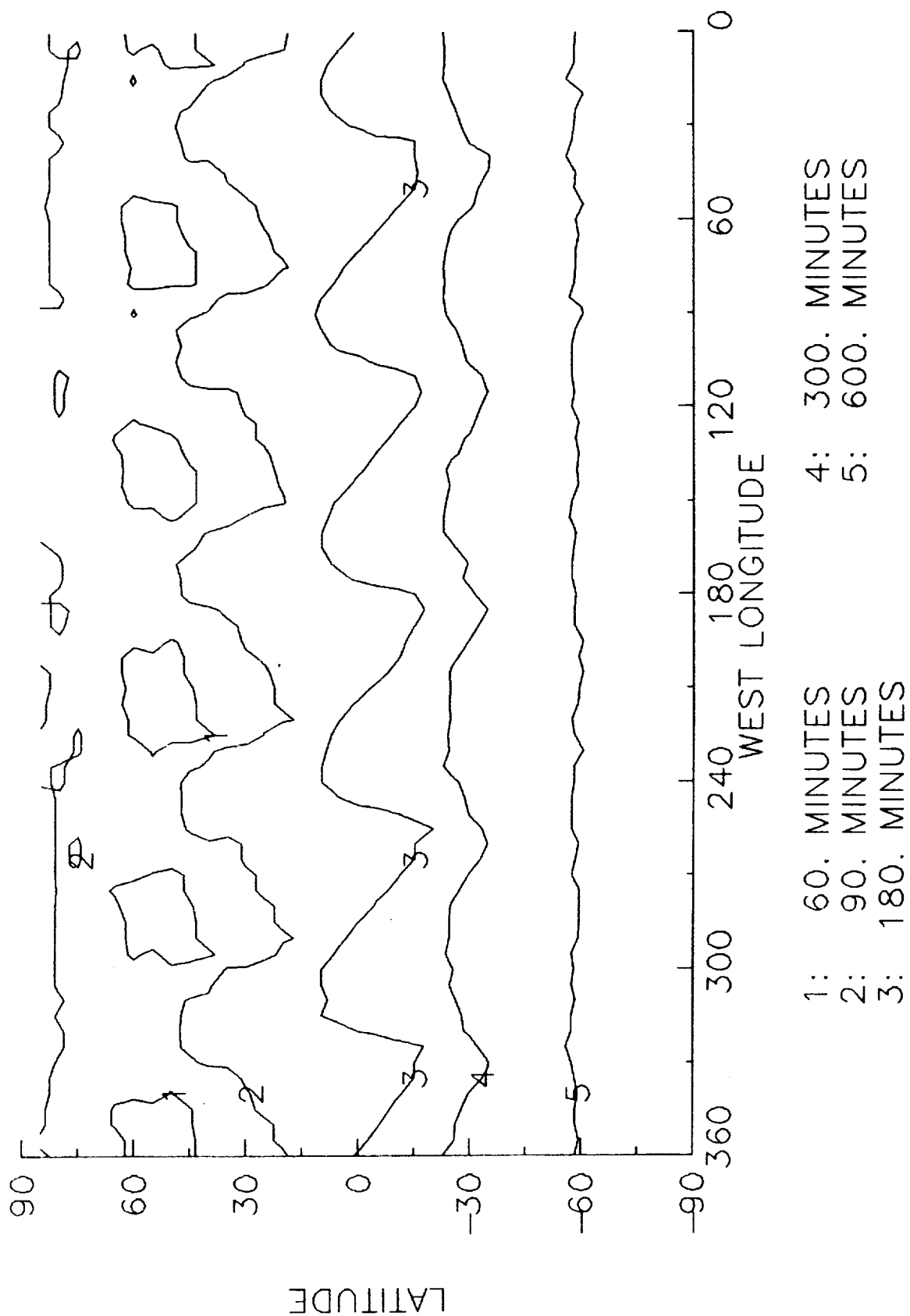
1: 180. MINUTES 3: 300. MINUTES
 2: 270. MINUTES 4: 320. MINUTES

COMM AVAIL. PER SOL, SOL #318 (8/20/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 45DEG ELV LIM
 (PERIAPSIS AT EQUATOR)

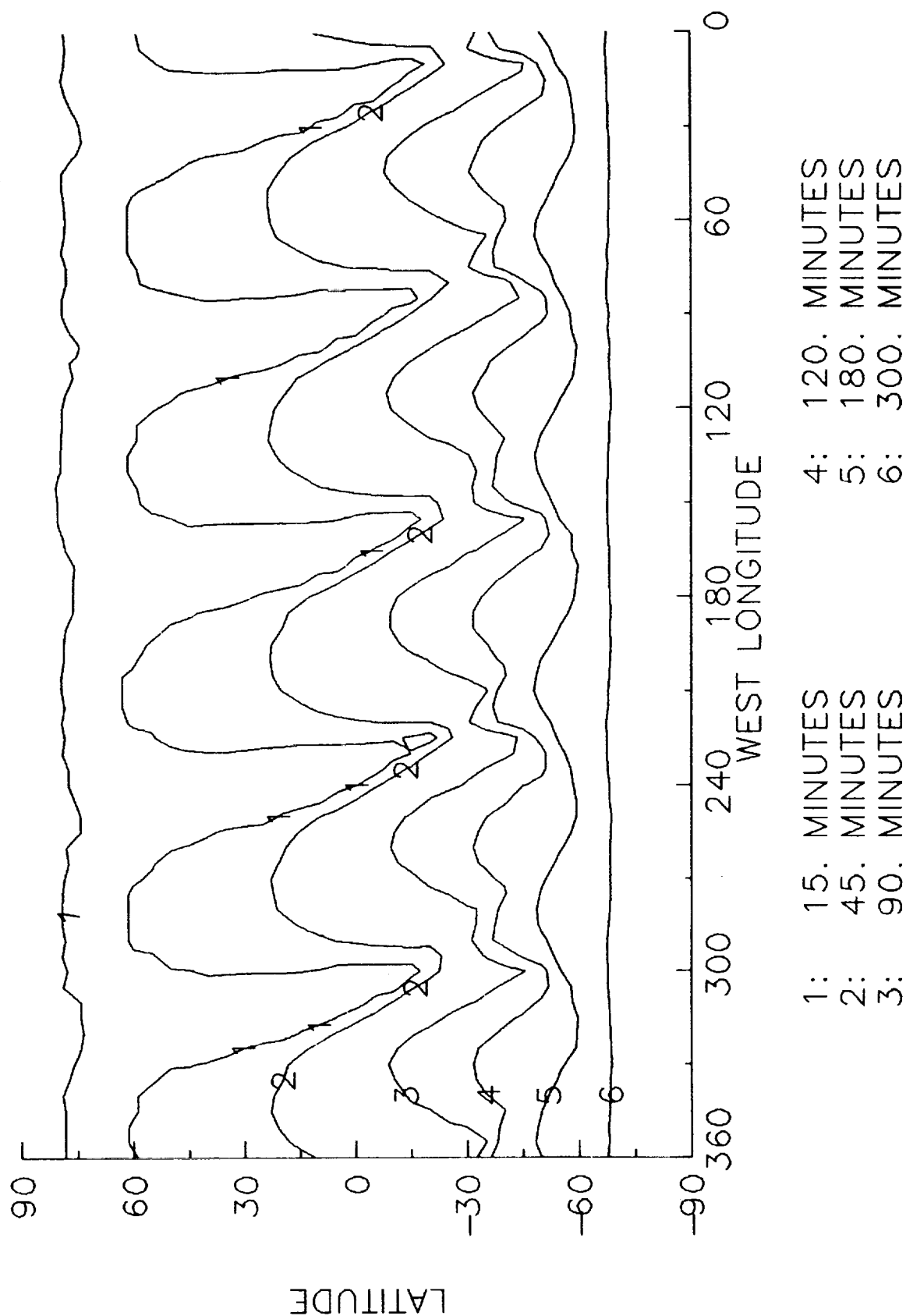


1: 30. MINUTES 3: 90. MINUTES
 2: 60. MINUTES 4: 120. MINUTES

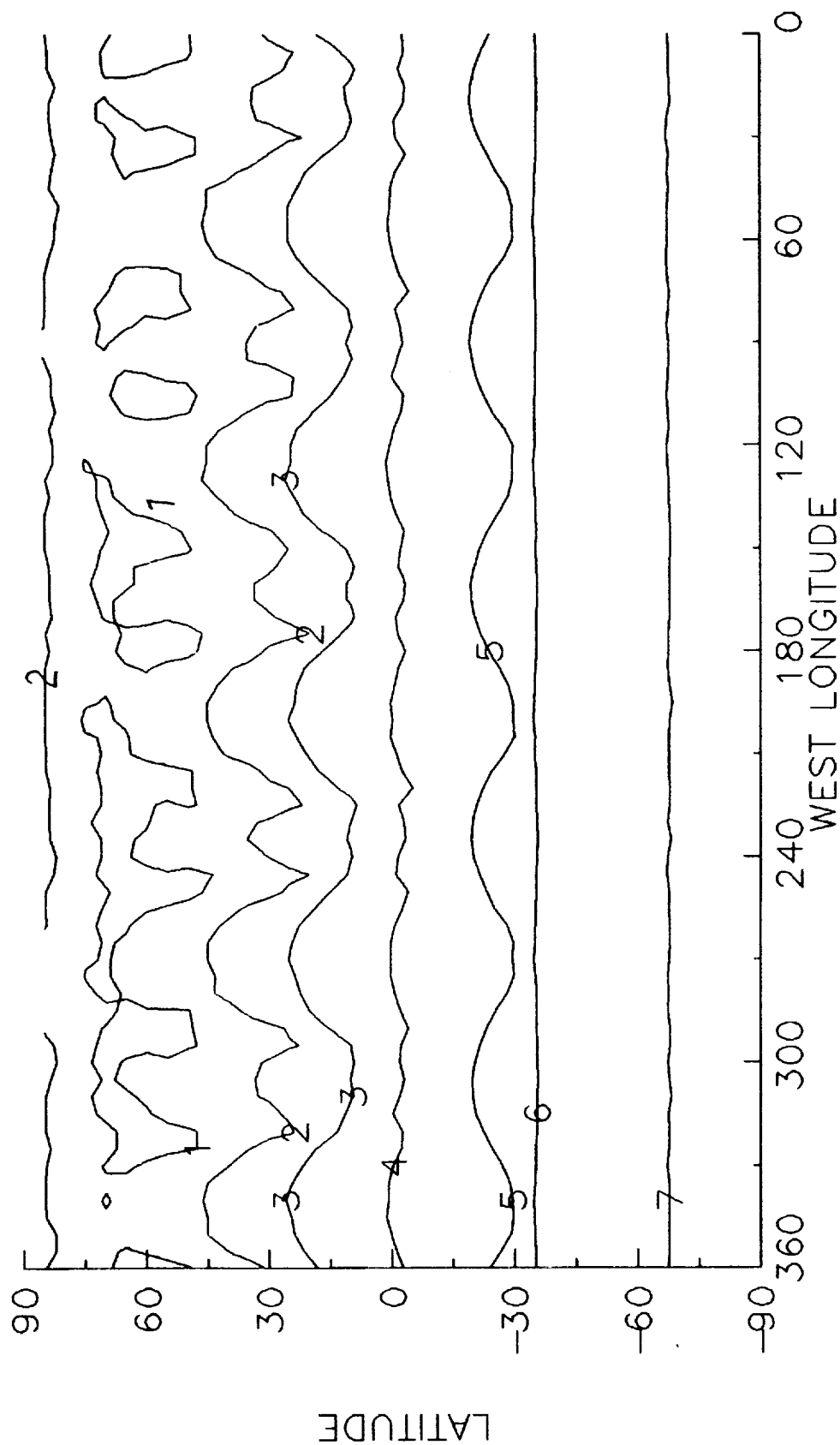
COMM AVAIL. PER SOL, SOL #361 (10/3/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 45DEG ELV LIM
 (PERIAPSIS AT 45°N)



COMM AVAIL. PER SOL, SOL #361 (10/3/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 45DEG ELV LIM
 (PERIAPS AT 45°N)

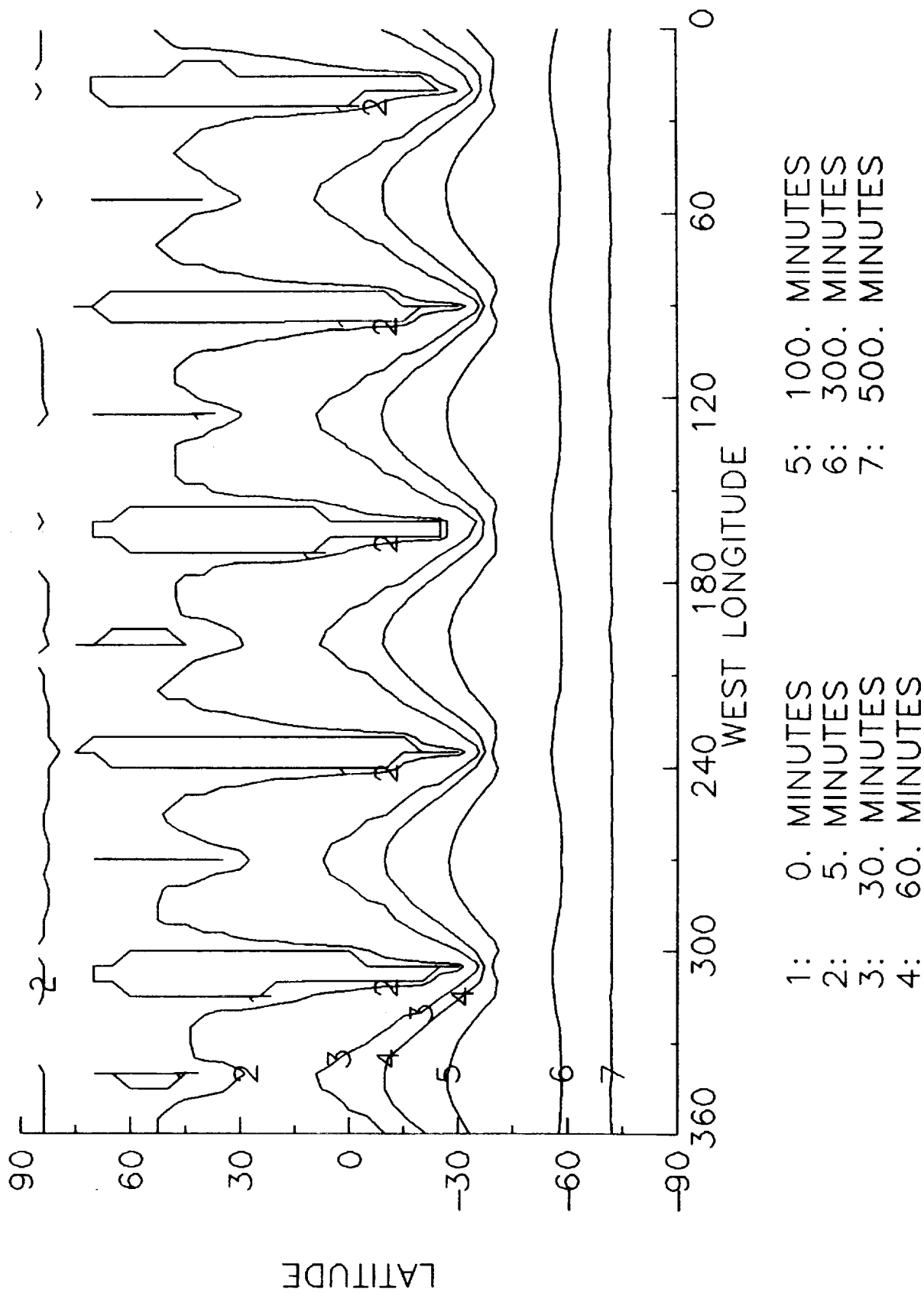


COMM AVAIL PER SOL, SOL #404 (11/16/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 15DEG ELV LIM
 (PERIAPSIS AT 90°N)



- 1: 15. MINUTES
- 2: 30. MINUTES
- 3: 60. MINUTES
- 4: 120. MINUTES
- 5: 300. MINUTES
- 6: 500. MINUTES
- 7: 800. MINUTES

COMM AVAIL PER SOL, SOL #404 (11/16/2000)
 HP=275KM, P=1/5 SOL, I=90DEG, 45DEG ELV LIM
 (PERIAPSIS AT 90°N)



COMMAP - COMMUNICATION AVAILABILITY MAP

SOL #404 15 deg

COMMUNICATIONS HORIZON ELEVATION ANGLE IS 15.0 DEGREES

ORBITAL ELEMENTS FOR THE ORBITER ON 19990929.0000:

A= 6986.2 E=0.474362 I= 90.00 NODE= 129.93 ARGP= 153.27 MEAN0= 0.00

START DATE: 20001116.0000 STOP DATE: 20001117.0260 TIME STEP IS 0.0010 DAYS

NUMBER OF MINUTES THAT THE SITE-ORBITER LINK IS AVAILABLE DURING THIS PERIOD TABULATED BELOW:

EAST LONGITUDE	LATITUDE																	WEST LONGITUDE
	-80.	-70.	-60.	-50.	-40.	-30.	-20.	-10.	0.	10.	20.	30.	40.	50.	60.	70.	80.	
0.	825.	791.	739.	651.	498.	315.	163.	128.	95.	68.	45.	24.	19.	13.	13.	19.	30.	360.
5.	827.	789.	740.	652.	498.	274.	167.	132.	101.	75.	53.	36.	26.	17.	14.	16.	33.	355.
10.	822.	792.	742.	660.	501.	233.	170.	135.	105.	81.	58.	40.	30.	22.	17.	17.	33.	350.
15.	822.	792.	739.	661.	511.	204.	171.	138.	109.	82.	59.	45.	30.	23.	17.	16.	30.	345.
20.	824.	791.	742.	664.	516.	204.	168.	137.	108.	82.	62.	43.	32.	22.	17.	17.	30.	340.
25.	824.	791.	740.	657.	501.	210.	170.	138.	108.	79.	59.	42.	30.	22.	16.	16.	30.	335.
30.	822.	792.	737.	655.	498.	265.	167.	135.	104.	75.	55.	39.	26.	20.	14.	17.	30.	330.
35.	825.	791.	736.	652.	498.	310.	164.	130.	96.	68.	45.	30.	16.	13.	13.	17.	32.	325.
40.	824.	791.	734.	648.	492.	333.	194.	137.	86.	56.	27.	22.	16.	13.	13.	17.	30.	320.
45.	821.	791.	739.	648.	492.	347.	236.	143.	94.	46.	33.	22.	19.	17.	14.	17.	30.	315.
50.	821.	788.	734.	642.	492.	351.	255.	141.	71.	50.	37.	29.	22.	16.	14.	16.	30.	310.
55.	822.	791.	733.	644.	491.	356.	258.	145.	71.	52.	39.	29.	23.	19.	16.	17.	32.	305.
60.	822.	793.	733.	644.	492.	350.	249.	138.	68.	49.	36.	27.	20.	16.	16.	16.	32.	300.
65.	822.	791.	734.	647.	492.	340.	228.	135.	84.	37.	29.	22.	16.	13.	13.	17.	33.	295.
70.	825.	788.	736.	649.	495.	324.	183.	124.	91.	62.	32.	23.	17.	13.	10.	17.	29.	290.
75.	825.	792.	736.	652.	494.	295.	163.	130.	98.	71.	49.	33.	22.	14.	13.	17.	29.	285.
80.	822.	791.	737.	657.	495.	240.	167.	134.	104.	78.	55.	39.	29.	20.	16.	17.	29.	280.
85.	822.	792.	737.	660.	500.	206.	168.	137.	107.	79.	58.	43.	27.	22.	17.	14.	29.	275.
90.	822.	791.	740.	660.	516.	203.	168.	137.	107.	81.	60.	42.	30.	23.	17.	16.	29.	270.
95.	824.	789.	740.	658.	504.	206.	170.	135.	108.	79.	59.	43.	30.	20.	16.	17.	29.	265.
100.	824.	791.	739.	654.	500.	236.	167.	134.	104.	78.	55.	39.	26.	20.	14.	17.	30.	260.
105.	822.	792.	737.	652.	498.	298.	164.	128.	98.	71.	48.	33.	20.	16.	13.	17.	30.	255.
110.	824.	791.	734.	648.	497.	325.	190.	124.	91.	60.	33.	22.	16.	12.	12.	17.	30.	250.
115.	822.	792.	734.	645.	495.	340.	222.	140.	91.	35.	26.	20.	14.	14.	14.	17.	33.	245.
120.	822.	792.	734.	642.	494.	351.	248.	140.	78.	50.	37.	27.	22.	19.	14.	17.	32.	240.
125.	824.	789.	734.	642.	491.	354.	253.	140.	71.	53.	39.	30.	24.	17.	14.	17.	32.	235.
130.	824.	792.	734.	642.	492.	351.	249.	134.	72.	53.	37.	29.	22.	19.	14.	16.	32.	230.
135.	822.	791.	734.	644.	497.	346.	236.	138.	84.	42.	27.	22.	16.	16.	14.	17.	32.	225.
140.	822.	789.	734.	648.	495.	333.	190.	120.	85.	55.	30.	23.	17.	13.	10.	17.	32.	220.
145.	821.	791.	736.	651.	497.	308.	163.	128.	95.	68.	45.	24.	17.	13.	12.	16.	30.	215.
150.	825.	789.	736.	655.	501.	246.	167.	132.	102.	75.	52.	37.	24.	19.	14.	17.	30.	210.
155.	825.	791.	740.	662.	500.	223.	170.	135.	107.	78.	56.	42.	29.	20.	14.	16.	30.	205.
160.	824.	791.	739.	661.	514.	204.	170.	135.	105.	81.	60.	42.	30.	23.	16.	13.	30.	200.
165.	822.	789.	737.	661.	510.	206.	170.	137.	107.	81.	58.	43.	30.	22.	17.	16.	30.	195.
170.	822.	789.	739.	657.	503.	228.	168.	134.	105.	79.	56.	39.	27.	20.	17.	19.	30.	190.
175.	822.	792.	737.	655.	498.	281.	166.	130.	101.	73.	50.	35.	24.	17.	16.	19.	30.	185.
180.	822.	791.	737.	651.	497.	318.	176.	125.	95.	65.	40.	22.	17.	12.	13.	19.	32.	180.
185.	824.	791.	736.	649.	497.	338.	199.	138.	84.	52.	26.	20.	16.	12.	14.	17.	30.	175.
190.	824.	791.	734.	645.	495.	350.	245.	141.	88.	46.	33.	24.	19.	16.	16.	19.	32.	170.
195.	825.	789.	734.	648.	494.	356.	255.	135.	71.	53.	37.	29.	22.	19.	16.	16.	32.	165.

EAST LONGITUDE	LATITUDE																WEST LONGITUDE	
	-80.	-70.	-60.	-50.	-40.	-30.	-20.	-10.	0.	10.	20.	30.	40.	50.	60.	70.	80.	
200.	822.	791.	736.	645.	494.	354.	258.	140.	72.	53.	39.	29.	23.	17.	14.	16.	32.	160.
205.	824.	789.	736.	645.	495.	351.	246.	140.	84.	48.	35.	23.	19.	17.	14.	16.	32.	155.
210.	822.	789.	736.	649.	498.	340.	219.	137.	82.	45.	30.	22.	17.	13.	13.	17.	30.	150.
215.	822.	791.	736.	648.	497.	321.	176.	127.	95.	65.	42.	24.	17.	13.	12.	17.	30.	145.
220.	824.	791.	737.	657.	500.	285.	168.	132.	101.	73.	52.	35.	23.	17.	14.	19.	30.	140.
225.	822.	791.	739.	660.	500.	239.	171.	137.	105.	79.	58.	42.	27.	20.	16.	16.	32.	135.
230.	824.	791.	740.	662.	507.	206.	170.	137.	108.	82.	62.	42.	32.	22.	17.	16.	30.	130.
235.	822.	792.	742.	661.	518.	207.	171.	138.	109.	82.	59.	43.	30.	23.	17.	14.	33.	125.
240.	824.	792.	739.	660.	505.	206.	170.	137.	107.	81.	59.	42.	30.	22.	17.	17.	33.	120.
245.	825.	791.	739.	657.	504.	243.	168.	134.	104.	76.	56.	37.	27.	19.	17.	19.	32.	115.
250.	824.	791.	737.	655.	503.	310.	164.	128.	98.	71.	48.	30.	19.	13.	14.	17.	32.	110.
255.	821.	792.	736.	654.	500.	333.	193.	131.	88.	59.	27.	22.	17.	12.	13.	19.	32.	105.
260.	824.	791.	737.	651.	500.	347.	235.	145.	94.	43.	29.	20.	16.	16.	14.	19.	32.	100.
265.	822.	791.	736.	647.	498.	356.	253.	143.	71.	52.	37.	30.	22.	19.	16.	16.	30.	95.
270.	824.	792.	736.	647.	497.	360.	259.	148.	73.	53.	39.	30.	23.	19.	17.	16.	29.	90.
275.	824.	793.	736.	645.	497.	356.	253.	140.	71.	52.	39.	29.	23.	17.	14.	17.	30.	85.
280.	824.	791.	736.	647.	497.	347.	235.	140.	88.	37.	29.	22.	17.	12.	14.	17.	30.	80.
285.	824.	789.	734.	651.	498.	333.	189.	122.	89.	60.	32.	23.	17.	13.	12.	17.	30.	75.
290.	824.	792.	739.	652.	498.	307.	166.	130.	98.	71.	49.	29.	19.	13.	14.	19.	30.	70.
295.	825.	792.	740.	655.	500.	248.	168.	132.	102.	78.	56.	39.	27.	19.	16.	19.	32.	65.
300.	824.	792.	743.	658.	505.	219.	170.	137.	107.	81.	60.	42.	30.	22.	17.	19.	33.	60.
305.	824.	792.	740.	661.	514.	207.	171.	140.	109.	82.	60.	45.	32.	22.	17.	16.	32.	55.
310.	822.	795.	742.	660.	507.	206.	171.	137.	107.	82.	60.	42.	30.	22.	17.	17.	35.	50.
315.	825.	791.	739.	658.	501.	232.	170.	137.	105.	78.	56.	40.	30.	19.	16.	19.	32.	45.
320.	824.	792.	737.	658.	500.	291.	167.	134.	99.	73.	52.	33.	23.	17.	14.	19.	32.	40.
325.	822.	791.	739.	651.	497.	323.	186.	125.	94.	63.	40.	22.	16.	12.	12.	17.	32.	35.
330.	822.	792.	737.	647.	497.	343.	216.	143.	92.	48.	27.	20.	16.	13.	13.	17.	32.	30.
335.	822.	792.	736.	647.	495.	351.	248.	145.	89.	49.	36.	29.	22.	16.	14.	16.	32.	25.
340.	825.	789.	737.	647.	494.	356.	259.	137.	73.	53.	42.	29.	23.	19.	17.	16.	32.	20.
345.	824.	789.	737.	645.	492.	354.	256.	132.	72.	53.	40.	29.	23.	19.	16.	16.	32.	15.
350.	825.	791.	737.	644.	495.	350.	245.	138.	85.	46.	32.	22.	17.	16.	14.	19.	30.	10.
355.	822.	792.	736.	648.	497.	338.	210.	132.	84.	52.	29.	23.	17.	13.	13.	17.	30.	5.
360.	825.	791.	739.	651.	498.	315.	163.	128.	95.	68.	45.	24.	19.	13.	13.	19.	30.	0.

COMMAP - COMMUNICATION AVAILABILITY MAP

SOL #404 45 deg

COMMUNICATIONS HORIZON ELEVATION ANGLE IS 45.0 DEGREES

ORBITAL ELEMENTS FOR THE ORBITER ON 19990929.0000:

A= 6986.2 E=0.474362 I= 90.00 NODE= 129.93 ARGP= 153.27 MEAN0= 0.00

START DATE: 20001116.0000 STOP DATE: 20001117.0260 TIME STEP IS 0.0010 DAYS

NUMBER OF MINUTES THAT THE SITE-ORBITER LINK IS AVAILABLE DURING THIS PERIOD TABULATED BELOW:

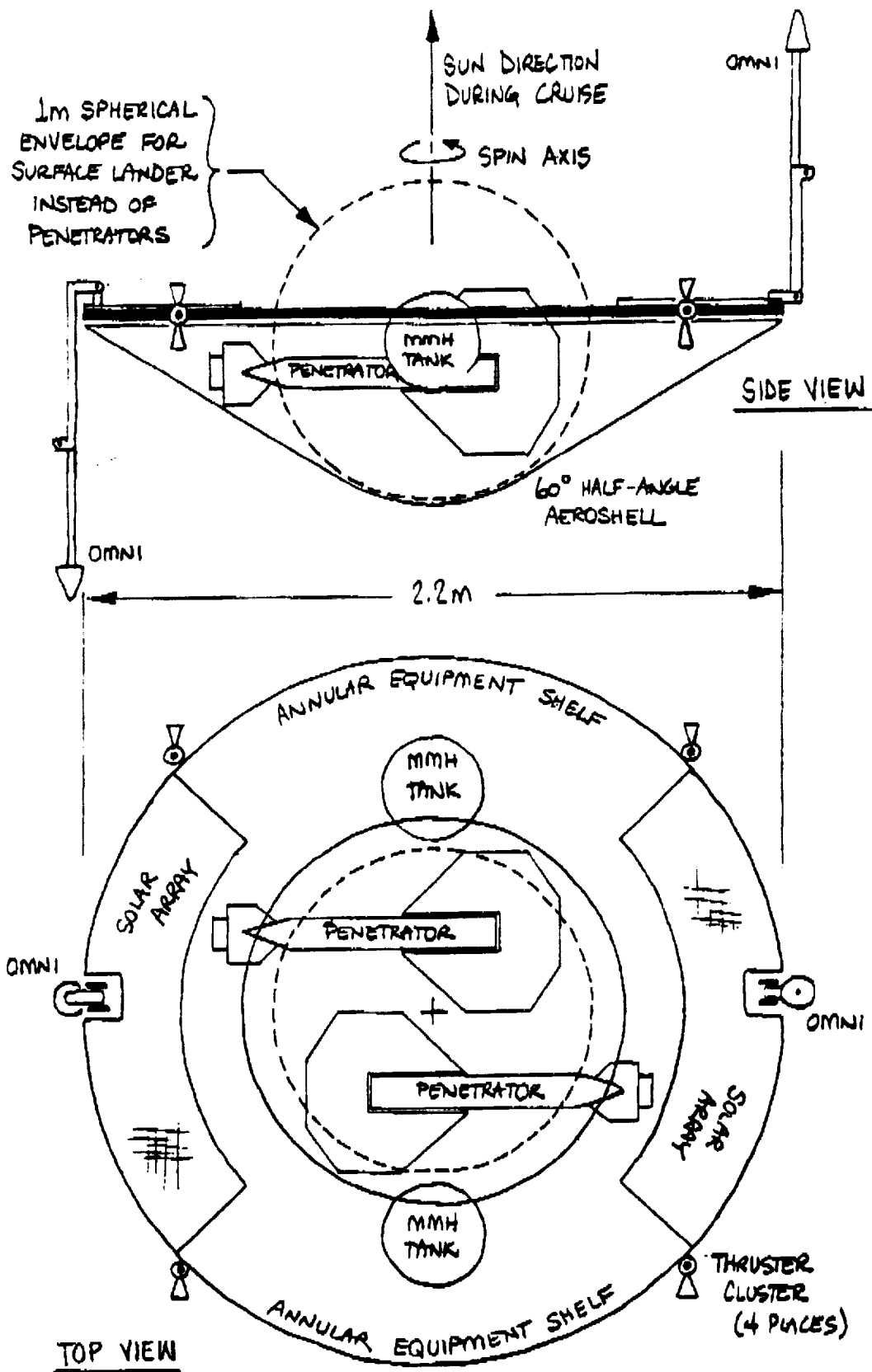
EAST LONGITUDE	LATITUDE																WEST LONGITUDE
	-80.	-70.	-60.	-50.	-40.	-30.	-20.	-10.	0.	10.	20.	30.	40.	50.	60.	70.	80.
0.	520.	441.	233.	134.	107.	75.	33.	26.	20.	14.	10.	9.	6.	4.	3.	3.	6.
5.	521.	438.	219.	141.	112.	85.	58.	27.	22.	16.	12.	9.	6.	4.	3.	3.	6.
10.	521.	438.	213.	144.	118.	92.	66.	45.	22.	16.	12.	7.	6.	3.	3.	3.	6.
15.	524.	438.	204.	145.	121.	95.	72.	52.	33.	19.	9.	4.	3.	0.	1.	1.	6.
20.	520.	436.	215.	145.	120.	95.	73.	52.	36.	22.	7.	3.	0.	0.	0.	1.	6.
25.	521.	439.	209.	143.	117.	94.	68.	49.	26.	13.	9.	6.	4.	3.	3.	3.	4.
30.	523.	439.	216.	140.	114.	86.	59.	30.	20.	14.	12.	9.	4.	4.	3.	3.	4.
35.	520.	441.	220.	135.	107.	76.	42.	24.	17.	14.	10.	7.	4.	4.	3.	3.	4.
40.	520.	439.	264.	128.	96.	59.	26.	20.	14.	10.	7.	6.	4.	3.	3.	3.	4.
45.	521.	441.	279.	143.	81.	27.	17.	12.	7.	3.	0.	0.	0.	0.	3.	3.	6.
50.	520.	439.	288.	170.	53.	10.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.	6.
55.	523.	441.	289.	177.	19.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.	6.
60.	520.	439.	287.	147.	63.	23.	12.	4.	0.	0.	0.	0.	0.	0.	1.	1.	4.
65.	520.	436.	275.	147.	86.	35.	24.	19.	13.	10.	7.	4.	3.	3.	3.	3.	6.
70.	518.	436.	252.	134.	99.	66.	32.	24.	17.	14.	10.	7.	6.	4.	3.	3.	6.
75.	521.	438.	220.	140.	109.	81.	50.	26.	20.	16.	12.	9.	6.	4.	3.	3.	7.
80.	520.	436.	215.	144.	114.	88.	62.	39.	22.	16.	12.	9.	6.	3.	3.	3.	7.
85.	521.	436.	209.	147.	117.	94.	69.	50.	30.	14.	9.	6.	3.	3.	1.	1.	6.
90.	518.	438.	215.	145.	120.	94.	71.	50.	33.	20.	9.	0.	0.	0.	0.	1.	6.
95.	518.	435.	206.	145.	118.	91.	68.	48.	30.	13.	9.	6.	3.	3.	1.	3.	6.
100.	518.	436.	213.	143.	115.	86.	63.	39.	20.	14.	10.	7.	6.	3.	3.	3.	6.
105.	520.	438.	219.	140.	108.	79.	49.	24.	19.	14.	10.	7.	6.	3.	3.	3.	6.
110.	520.	438.	255.	131.	99.	65.	29.	22.	16.	13.	9.	7.	6.	3.	3.	3.	7.
115.	523.	439.	276.	138.	85.	30.	22.	14.	10.	7.	6.	4.	3.	3.	3.	3.	7.
120.	520.	439.	287.	147.	65.	19.	7.	0.	0.	0.	0.	0.	0.	0.	0.	1.	7.
125.	520.	442.	292.	179.	14.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	9.
130.	518.	439.	287.	170.	50.	16.	0.	0.	0.	0.	0.	0.	0.	0.	0.	3.	7.
135.	520.	441.	279.	148.	78.	30.	20.	14.	10.	7.	4.	1.	3.	3.	3.	3.	7.
140.	523.	441.	268.	130.	94.	58.	29.	22.	17.	12.	9.	7.	4.	3.	3.	3.	7.
145.	520.	438.	223.	137.	107.	75.	36.	26.	19.	16.	12.	9.	6.	4.	3.	3.	6.
150.	520.	439.	219.	143.	114.	86.	58.	27.	22.	16.	12.	9.	6.	4.	3.	3.	6.
155.	523.	436.	213.	145.	117.	91.	68.	46.	24.	14.	10.	7.	4.	3.	3.	3.	4.
160.	521.	438.	215.	147.	120.	94.	71.	50.	35.	20.	9.	3.	0.	0.	0.	0.	6.
165.	523.	438.	212.	147.	118.	94.	69.	50.	32.	19.	7.	4.	1.	0.	0.	1.	6.
170.	523.	441.	210.	144.	115.	91.	66.	45.	20.	14.	10.	7.	6.	3.	1.	1.	7.
175.	520.	439.	217.	141.	111.	84.	56.	26.	19.	14.	10.	7.	6.	4.	3.	3.	7.
180.	521.	439.	238.	135.	102.	72.	30.	23.	17.	13.	10.	7.	6.	4.	3.	3.	7.
185.	523.	439.	271.	128.	92.	50.	24.	17.	13.	9.	7.	6.	4.	3.	3.	3.	4.
190.	521.	439.	284.	148.	73.	24.	16.	9.	3.	0.	0.	0.	0.	0.	1.	1.	4.
195.	521.	442.	289.	177.	42.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	1.	4.

Session B, Submittal No. 4

Alan L. Friedlander
Science Applications International Corporation

MARS NETWORK LANDER (FREE-FLYER CONCEPT)

**ALAN FRIEDLANDER
SAIC**



MARS NETWORK LANDER(S)

(FREE-FLYER CONCEPT)

MARS NETWORK LANDER (FREE-FLYER CONCEPT)

MASS STATEMENT

CRUISE DISC (DRY) _____ 91.8 KG

STRUCTURE	20.0	FROM PVO	} JETTISONED BEFORE ENTRY
NAVIGATION	8.8	STAR & SUN SENSORS	
COMMAND/CONTROL	18.4	FROM PVO	
COMMUNICATION	12.9	INCLUDES OMNI'S	
POWER CONDITIONING	7.3	USES LANDER BATTERIES	
SOLAR ARRAYS	6.0	90W @ MARS	
ATTITUDE CONTROL	6.4	FROM PVO	
PROPELLANT TANKS	2.0		
THERMAL CONTROL	10.0		

ENTRY SYSTEM _____ 43.2

AEROSHELL STRUCTURE	20.0
AEROSHELL ABLATOR	18.2
SEPARATION SYSTEM	2.0
PARACHUTE MOTOR	2.0
ENTRY INSTRUMENTATION	1.0

DRY MASS CONTINGENCY (15%) _____ 20.0

LANDER ALLOWANCE _____ 115.0

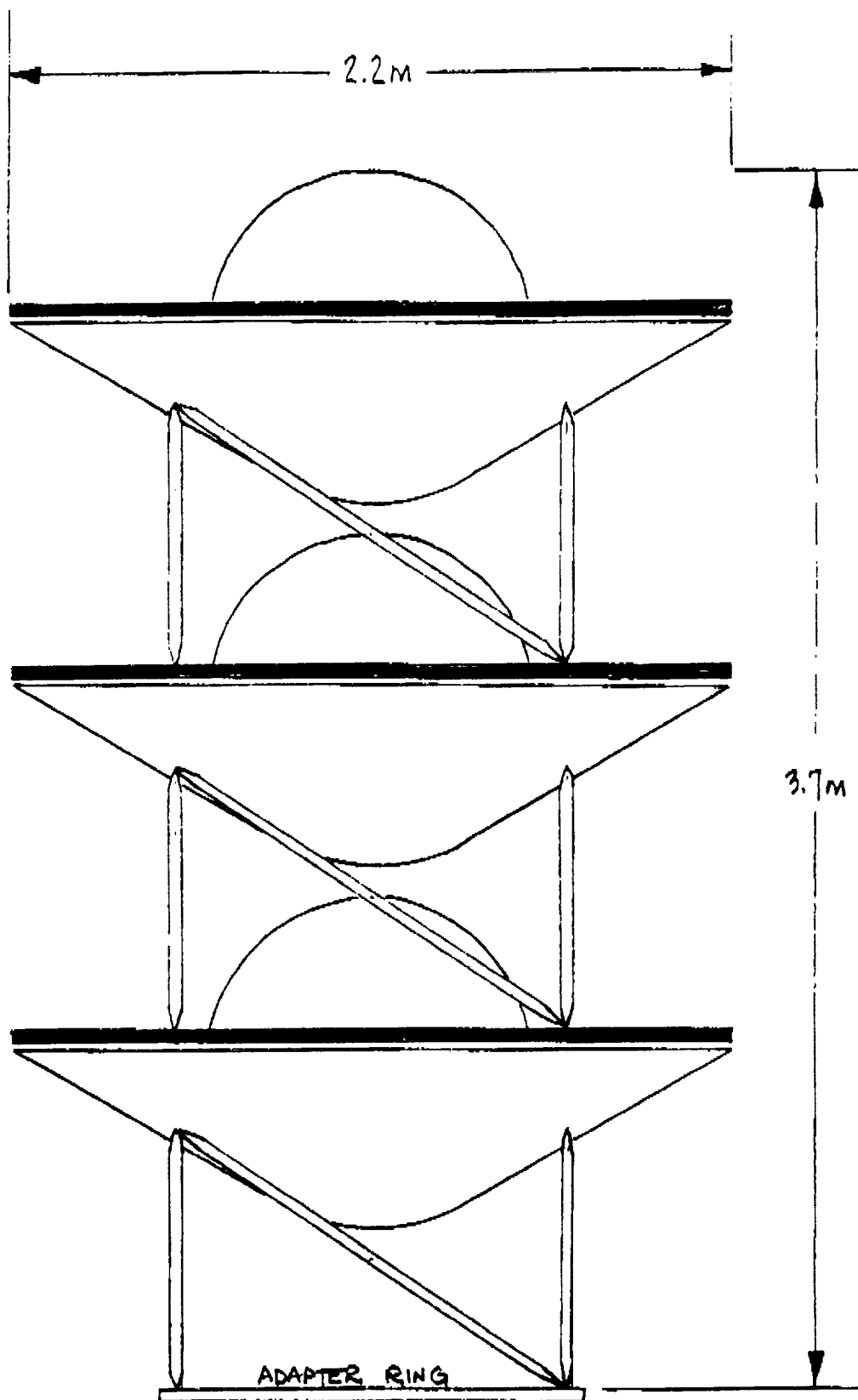
PROPELLANT (MMH MONOPROP FOR $\Delta V = 125$ M/SEC) _____ 15.0

TOTAL MASS _____ 285.0 KG

TRIPLE-STACK CONFIGURATION

3-LANDERS	855
INTERPROBE/RING ADAPTERS	95
TOTAL	950 KG

DELTA 7925 CAPABILITY @ $C3 = 15$ _____ 950 KG



MARS NETWORK LANDERS
TRIPLE-STACK LAUNCH CONFIGURATION
TOTAL MASS ~ 950 KG

Session B, Submittal No. 5

Alan L. Friedlander
Science Applications International Corporation

**ANALYSIS OF SUCCESS PROBABILITY/COST TRADES
FOR SMALL LANDERS IN A MARS NETWORK**

**ALAN FRIEDLANDER
SAIC**



With addendum of
new results

December 6, 1989

MEMO TO: Roger Bourke, EIS Team
FROM: Alan Friedlander, SAIC
SUBJECT: Analysis of Success Probability/Cost Trades for Small Landers in a Mars Network

The premise to be tested in this analysis is whether cost economies may accrue by delivering more landers designed to lower reliability of operation (compared to fewer landers of higher reliability) to obtain a desired probability of achieving a given number of lander successes. Generally, the application in mind is a network of penetrators, although the analysis may apply as well to other small lander concepts or even to simple rovers. In previous MRSR studies, the approach taken to raise the probability of a successful mission (e.g. a rover or sample return objective) was to invoke a dual launch policy utilizing identical flight systems. With this approach we found that a substantial improvement in achieving at least one total mission success was gained for realistic values of system element reliability, albeit at the expense of higher program cost and more complex operations. However, in the case of a large number of small landers whose recurring cost of production might be small compared to the development cost, a single spacecraft carrier may be sufficient to deliver these landers to Mars within acceptable limitations of spacecraft injected mass and launch vehicle performance capability. It seems reasonable to at least explore the question of potential economies if such landers were purposely designed to lower values of reliability. What is specifically meant by lower reliability in this context is that, while fewer lander emplacements will succeed, those that do succeed will accomplish the desired mission objectives. The underlying assumption here is a certain degree of independence of lander system failure modes such that objective-specific elements (science instruments and data communications) are highly reliable while delivery-specific elements (e.g. deorbit propulsion and aeroshell) are less reliable and developed at lower cost with attendant higher risk. This analysis leaves open the important question as to whether such an approach is at all realistic in terms of engineering design, but focuses instead on the first question of potential cost advantage.

The method of analysis is based on a probabilistic model of lander success and a related probabilistic model of project cost including the lander, spacecraft carrier, and integration, but not launch or operations costs. Quantitative results are obtained in a normalized and parametric fashion. Sensitivity to the assumed model parameters is also examined.

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Mission Success Model

Consider (n) landers each of which have the same level of reliability (p) for achieving individual mission success. Assuming that the actual failure events of different landers are statistically independent (even though the underlying failure modes for contributing components may be related), then the probability that exactly (m) of these landers are successful is given by the binomial distribution

$$P(m \text{ successes}) = \left[\frac{n!}{(m! * (n - m)!)} \right] * p^m * (1 - p)^{n-m} \quad (1)$$

where ! denotes the factorial operator and * denotes multiplication. Mission success also depends on the reliability of the launch vehicle and the spacecraft carrier that delivers the landers to Mars. To take these factors into account, we define P_L as the probability of a successful launch event and P_C as the probability of a successful delivery event. Then, the overall probability P that at least m (i. e. m or more) landers will be successful (for a single launch) is calculated by the expression

$$P = P_L P_C \sum_{i=m \text{ to } n} \left[\frac{n!}{(i! * (n - i)!)} \right] * p^i * (1 - p)^{n-i} \quad (2)$$

The relationships of Equation (2) are illustrated in Figure 1 for $P_L = 0.94$, $P_C = 0.98$, and $p = 0.8$.

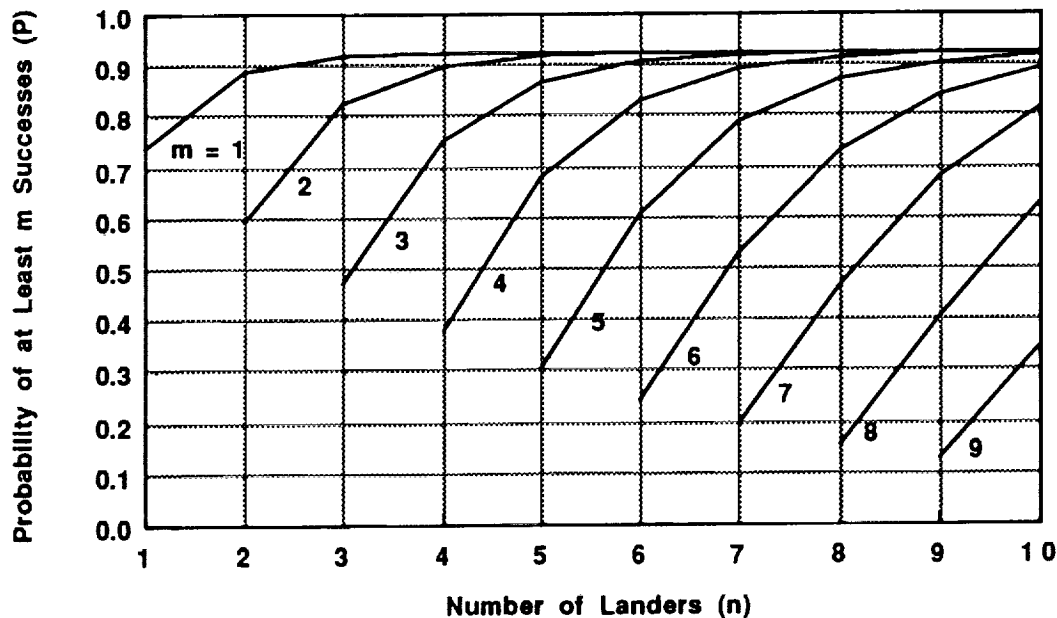


Figure 1 Lander Success Probability $P(n, m)$ for Lander Reliability $p = 0.8$
 $P_L = 0.94$, $P_C = 0.98$

Mission Cost Model

The lander system development cost is modeled in terms of design reliability by the relationship

$$C_d = C_{do} / (1 - p)^a \quad (3)$$

where C_{do} is a "reference" development cost at $p = 0$, and the exponent (a) is a model parameter. This equation is graphed in Figure 2 for values of $a = 0.1, 0.2$, and 0.301 .

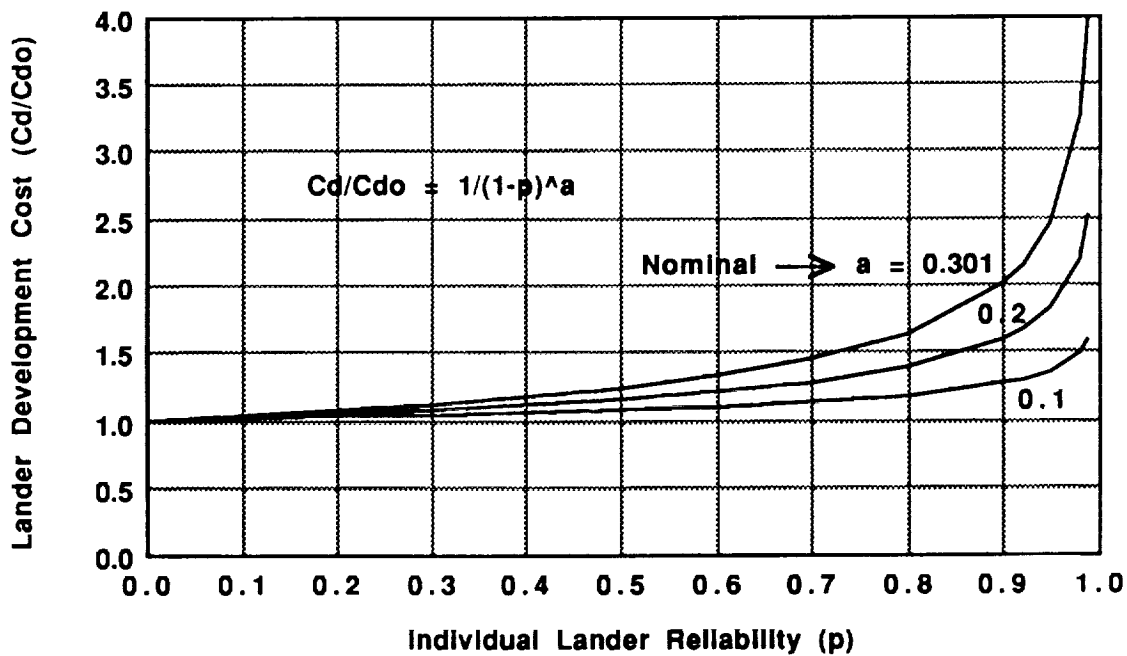


Figure 2 Lander Development Cost Model

The nominally selected value of the development cost parameter is $a = 0.301 = \log_{10} 2$, which gives a doubling of cost from $p = 0$ to $p = 0.9$ and doubling again for $p = 0.99$, etc. For $a = 0.1$ the increase in cost is only 25% for each additional 9 in reliability. The sensitivity to this parameter will be tested later. Recurring cost for each additional lander is assumed to be a constant fraction of the development cost. Hence, the lander system cost model is represented by

$$LC = C_d (1 + k_1 * n) = C_{do} (1 + k_1 * n) / (1 - p)^a \quad (4)$$

where the nominal value of the constant is selected as $k_1 = 0.2$. Total project cost includes the lander, carrier, and a cost element associated with hardware integration, management, and

contingency. The carrier spacecraft cost is taken to be proportional to the reference lander cost (development + recurring) at $p = 0$. Integration, management, and contingency is taken to be proportional to the sum of the carrier cost and the reference lander cost. Hence, the total project cost model is represented by

$$\begin{aligned} PC &= LC + k_2 * C_{do} (1 + k_1 * n) + k_3 * [LC + k_2 * C_{do} (1 + k_1 * n)] \\ &= C_{do} (1 + k_1 * n) [1 / (1 - p)^a + k_2 (1 + k_3) + k_3] \end{aligned} \quad (5)$$

where the nominal parameters are $a = 0.301$, $k_1 = 0.2$, $k_2 = 0.667$, and $k_3 = 0.4$. The final step in the cost model is to normalize LC and PC to their respective values LC^* and PC^* corresponding to one lander ($n = 1$) and reliability $p = 0.8684$ evaluated at the nominal values of the cost model parameters. Hence, $LC^* = 2.209 C_{do}$ and $PC^* = 3.809 C_{do}$. Lander system relative cost and total project relative cost are graphed in Figures 3 and 4 as a function of the number of landers and the individual lander reliability.

Results

Solution of the mission success model (Equation 2) was obtained for a constant probability $P = 0.8$ that at least (m) landers will be successful. These calculations assume the nominal values of 0.94 for launch success and 0.98 for carrier success; these values yield the reference lander reliability of $p = 0.8684$ for a single lander. Results are shown in Figure 5 which plots the required lander reliability as a function of the number of landers (n) and the minimum number of lander successes (m). The solution values for p are then used to evaluate the normalized total project cost which is graphed in Figure 6. Note that for each value of (m) there is a number of landers ($n > m$) that yields the lowest cost. Generally, (n) is greater than (m) by one or two lander units. This result substantiates the initial contention that more landers of lower reliability may provide cost economy. The intersection points along the minimum cost locus can be mapped into Figure 5 to determine the lander reliability values; the range is $p = \{0.64, 0.87\}$ as m varies from 1 to 8. For example, to obtain at least six lander successes ($m = 6$) at a probability of 80%, the minimum relative cost is $PC/PC^* = 2.084$ (i.e. twice the single lander project cost) with $n = 8$ and $p = 0.835$. Note also that the cost curve is fairly flat for $n > 8$, so that if $n = 10$ the project cost increases to only 2.195 but the required lander reliability decreases to $p = 0.711$. By comparison, if $n = m = 6$, then the required reliability is quite high at $p = 0.977$ and the project relative cost increases to 2.563. One could also interpret the results for a constant cost as (m) varies. For example, if $PC/PC^* = 2.0$ or less, then for values of $m = \{1, 2, 3, 4, 5\}$ the minimum necessary lander reliabilities are $\{0.18, 0.31, 0.42, 0.57, 0.74\}$ at corresponding values of $n = \{10, 10, 10, 9, 8\}$.

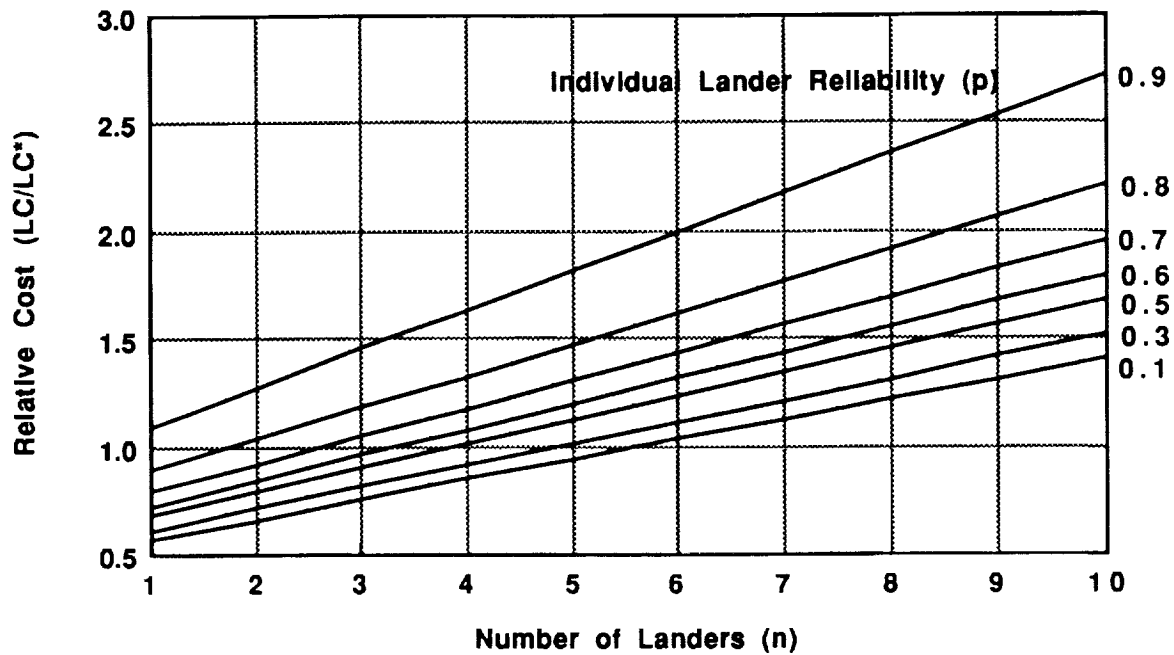


Figure 3 Lander System Relative Cost
 $LC^* = 2.209 \times C_{do}$ (for $n = 1$, $p = 0.8684$)

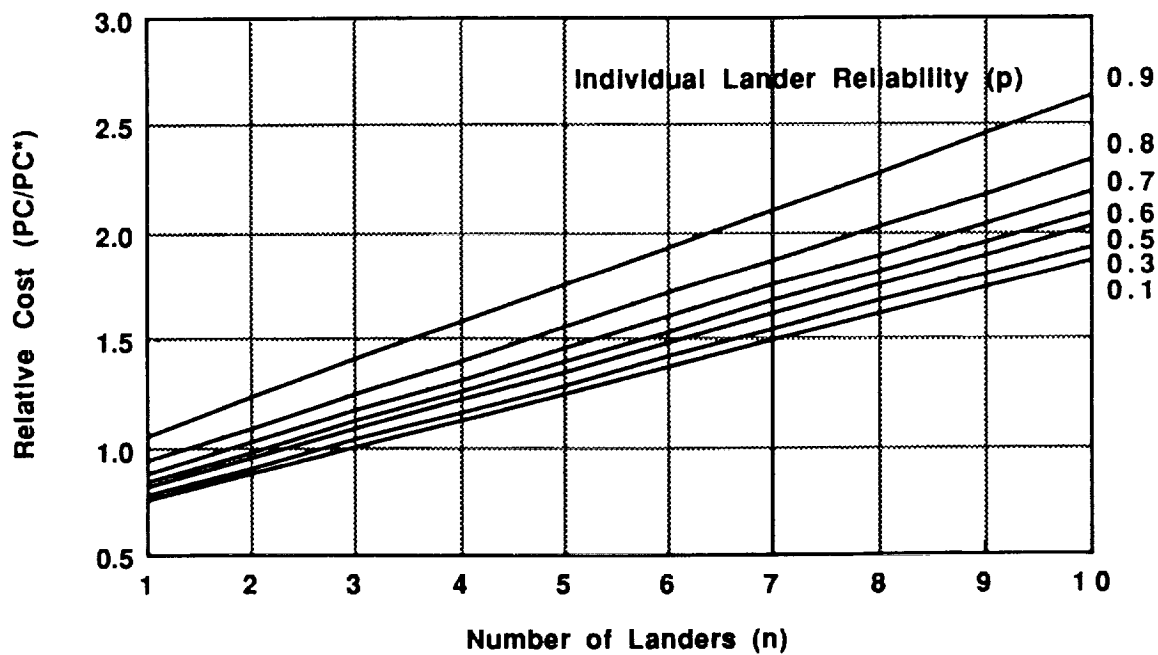


Figure 4 Total Project Relative Cost
 $PC^* = 3.809 \times C_{do}$ (for $n = 1$, $p = 0.8684$)

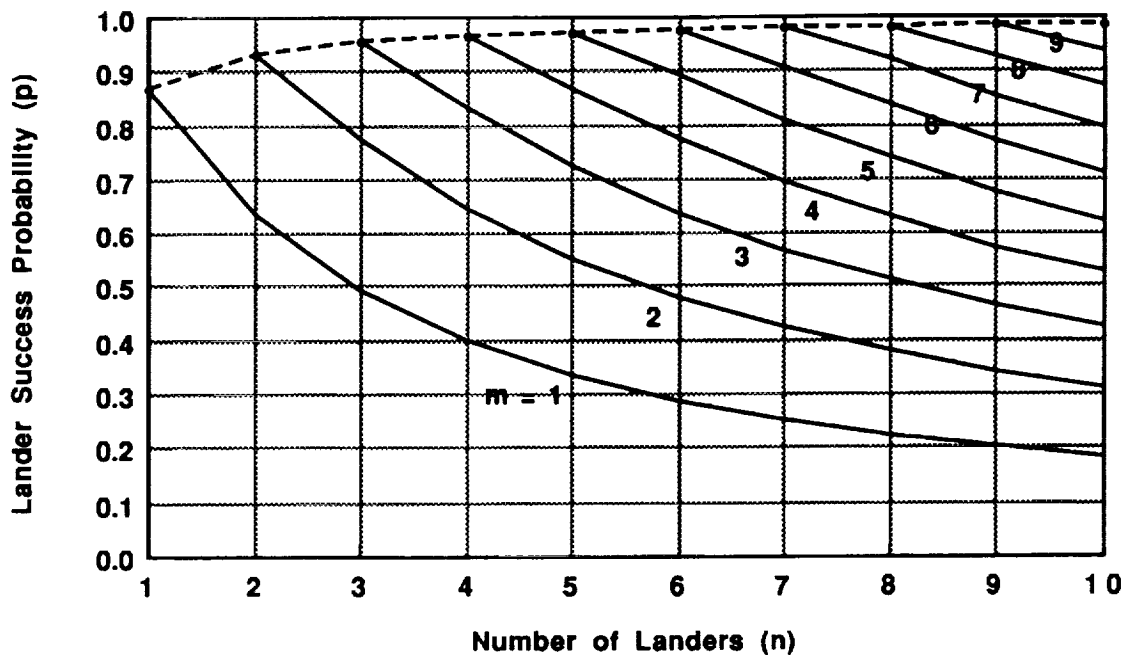


Figure 5 Required Lander Reliability for at Least m Lander Successes with Probability $P = 0.8$ ($P_l = 0.94$, $P_c = 0.98$)

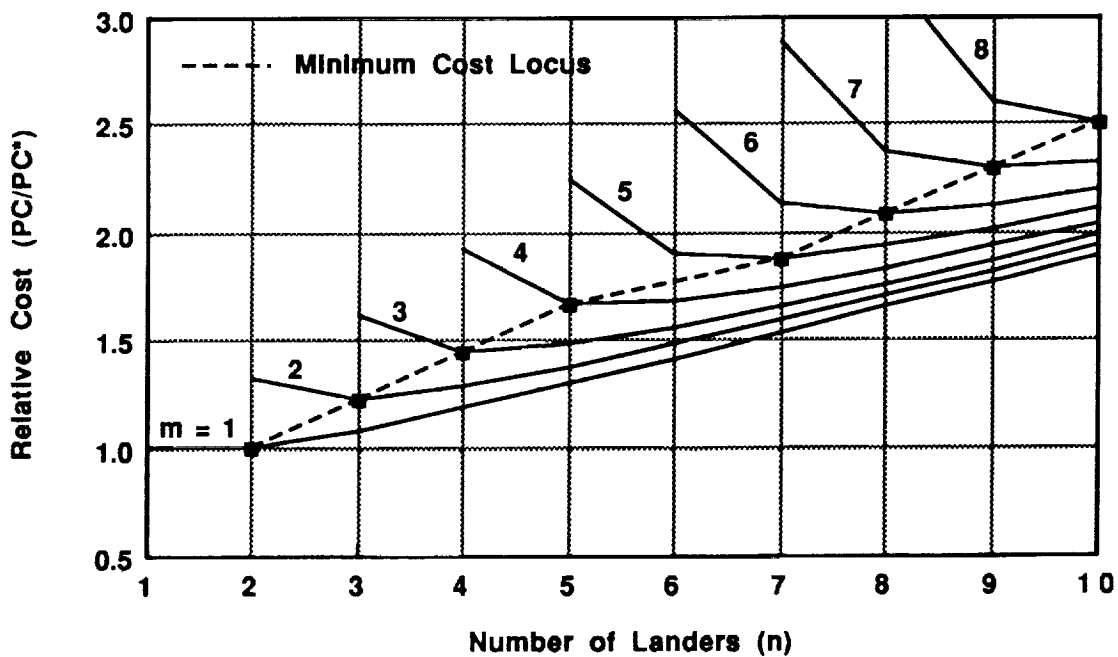


Figure 6 Project Cost for at Least m Lander Successes with Probability $P = 0.8$ ($P_l = 0.94$, $P_c = 0.98$)

Similar types of solution data are presented in Figures 7 and 8 for a constant success probability of $P = 0.9$. In this case, of course, the level of both cost and required reliability is raised to satisfy the more demanding 90% success capability. For example, at $m = 6$, the minimum relative cost is $PC/PC^* = 2.327$ obtained for $n = 9$ and $p = 0.866$. If $PC/PC^* = 2.0$ or less, then for values of $m = \{1, 2, 3, 4\}$ the minimum necessary lander reliabilities are $\{0.31, 0.45, 0.61, 0.76\}$ at corresponding values of $n = \{10, 10, 9, 8\}$.

Sensitivity to Model Parameters

Model parameters were varied, generally one at a time, to determine the sensitivity of the minimum PC/PC^* solution. These calculations were made for the case of $m = 6$ and $P = 0.8$ with PC^* held constant at its reference value $3.809 C_{d0}$. Results are listed in Table 1.

Table 1
Sensitivity to Model Parameters for $m = 6$ and $P = 0.8$
(Parameters at Nominal Values Unless Otherwise Noted)

Varied Parameter	n	p	PC/PC*
a = 0.1	6	0.9768	1.611
0.2	7	0.9074	1.854
0.301	8	0.8351	2.084
0.4	9	0.7694	2.302
k ₁ = 0.1	9	0.7694	1.441
0.2	8	0.8351	2.084
0.3	8	0.8351	2.725
k ₂ = 0.5	8	0.8351	1.925
0.667	8	0.8351	2.084
1.0	8	0.8351	2.402
k ₃ = 0.2	8	0.8351	1.857
0.4	8	0.8351	2.084
0.6	8	0.8351	2.312
P ₁ *P _C = 0.84	8	0.8910	2.240
0.9212	8	0.8351	2.084
1.00	8	0.8014	2.020
P ₁ *P _C = 0.84, a = 0.4 k ₁ = 0.3, k ₂ = 1.0, k ₃ = 0.6	9	0.8338	4.127
P ₁ *P _C = 1.0, k ₁ = k ₂ = 0	9	0.7324	1.093

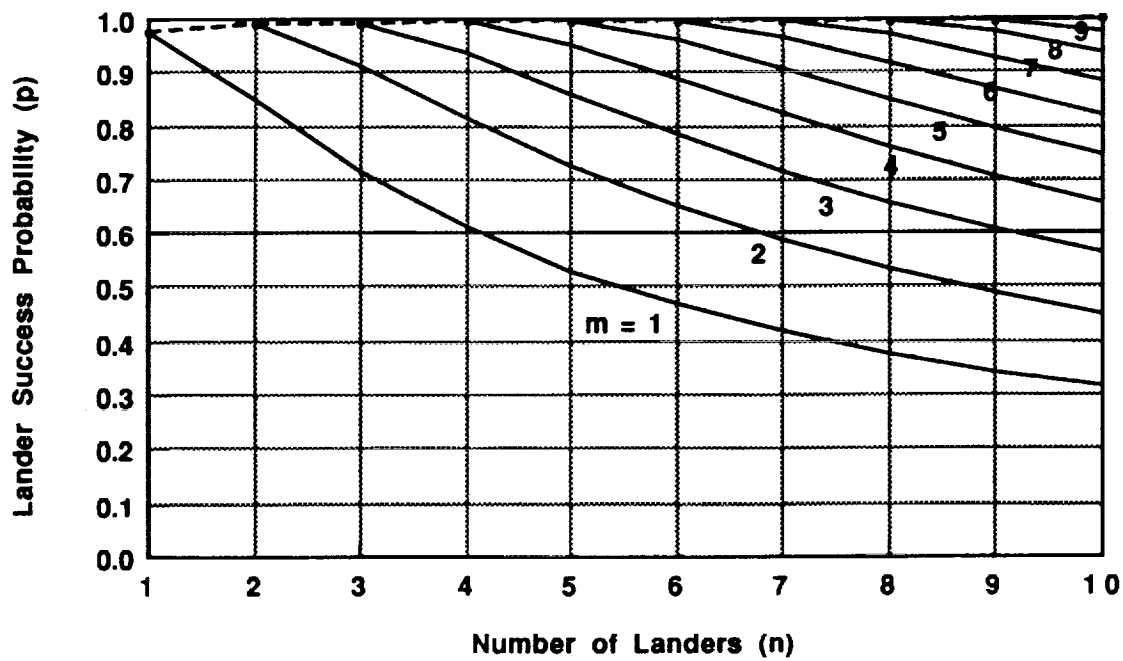


Figure 7 Required Lander Reliability for at Least m Lander Successes with Probability $P = 0.9$ ($P_I = 0.94$, $P_c = 0.98$)

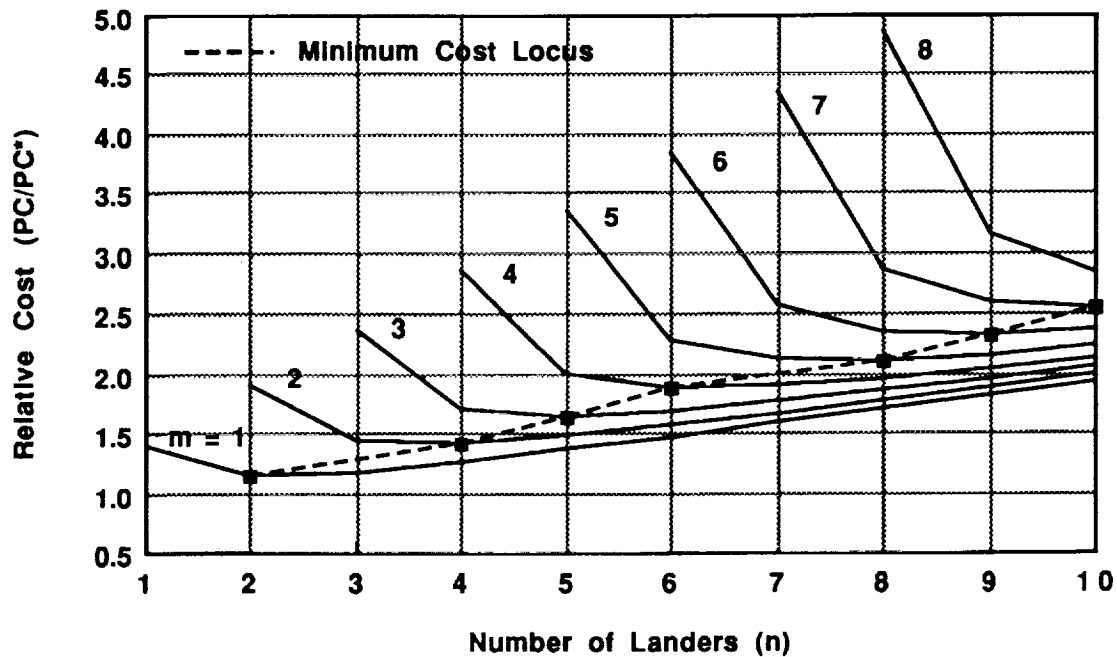


Figure 8 Project Cost for at Least m Lander Successes with Probability $P = 0.9$ ($P_I = 0.94$, $P_c = 0.98$)

Comparison of One and Two Launch Scenarios

Results presented so far have been for a single launch of (n) landers. Additional calculations were made for two launches, but this required modification of the mission success and cost models. To calculate the probability P for at least (m) lander successes with two launches, it is necessary to use Equation (1) as the basic model for lander success, multiply each term by the product $P_1 * P_c$ except for the $m = 0$ term which is adjusted to $[(1 - P_1 * P_c) + P_1 * P_c * P_{m=0}]$, and then obtain the various combinations for exactly (m) successes with two launches. The probability for at least (m) lander successes can then be calculated by summation of terms as in Equation (2). The project cost model for two launches is taken as a modification of Equation (5)

$$PC(2) = C_{do} \left\{ (1 + k_1 * n) \left[1 / (1 - p)^a + k_3 \right] + 2 * k_2 (1 + k_3) * (1 + k_1 * n/2) \right\} \quad (6)$$

where (n) is the total number of landers for two launches.

Employing the nominal values of model parameters, the first comparison case examined is $n = 4$ and a constant probability $P = 0.8$ that at least (m) landers will be successful. The single launch carries 4 landers while the dual launch system carries 2 landers each. Results for this case are listed in Table 2.

Table 2
Comparison of One and Two Launches for $n = 4$ and $P = 0.8$

At Least m Successes	<u>One Launch (n=4)</u>		<u>Two Launches (n = 2+2)</u>	
	<u>p</u>	<u>PC/PC*</u>	<u>p</u>	<u>PC/PC*</u>
1	0.3977	1.180	0.3675	1.417
2	0.6447	1.275	0.7747	1.615
3	0.8336	1.441	0.8951	1.806
4	0.9653	1.929	0.9854	2.561

Although the relative cost for two launches is always higher, if the criterion of comparison is the minimum value of lander reliability (p), then the results indicate that two launches is better only for the condition $m = 1$. If more than 2 lander successes is desired, a higher reliability is required because of the influence of possible launch and carrier failures.

The second comparison case examined is $n = 8$ and a constant value of $p = 0.8$ for the reliability of each lander. In this case we compare the mission success probability $P(m)$ for $m = 1$ to 8. The relative project costs are $PC(1)/PC^* = 2.018$ and $PC(2)/PC^* = 2.263$ for all values of (m) . Results are listed in Table 3.

Table 3
Comparison of One and Two Launches for $n = 8$ and $p = 0.8$

At Least m Successes	One Launch ($n = 8$) P	Two Launches ($n = 4+4$) P
1	0.9212	0.9936
2	0.9211	0.9898
3	0.9201	0.9665
4	0.9116	0.8992
5	0.8694	0.8008
6	0.7341	0.6763
7	0.4637	0.4271
8	0.1546	0.1424

These results indicate a "success performance" crossover point between one and two launches at the value $m = \{3, 4\}$. That is, two launches are better as measured by probability of success only for the condition $m = 1, 2$, or 3. If 4 or more lander successes is desired, then the single launch policy yields a somewhat higher probability of that occurrence.

Addendum to December 6, 1989, Memorandum

I did some more sensitivity studies relative to the cost model assumption. The results still confirm the conjecture (generally) that more landers at lower reliability yield lower project cost.

Basic Cost Model No.1 $C_d/C_{d_o} = \frac{1}{(1-p)^a}$
(as per memos)

Basic Cost Model No.2 $C_d/C_{d_o} = \left(\frac{1}{1-p}\right)^a$
(modified "Bourke")

Results for $m=6$ and $P(m \geq 6) = 0.8$

Cost Model	a	n	Minimum Cost Solution	
			p	PC/C_{d_o}
No. 1	0.100	6	0.9768	6.136
	0.301	8	0.8351	7.938
	0.500	9	0.7693	9.561
	1.000	13	0.5767	10.334
No. 2	0.100	6	0.9768	6.130
	0.301	8	0.8351	7.703
	0.500	10	0.7113	8.710
	1.000	15	0.5110	9.513

Note: The greater the sensitivity (a), the more landers (n) desired.

Combining Independent and "Common Cause" Failure Events

Consider (n) landers on a single launch. Each lander has an independent reliability = p_i and a common cause (or bias) reliability component = p_d . Then, if S_m represents the event of exactly m successes, the total conditional probability formula is

$$P(S_m) = P(S_m/\bar{D})P(\bar{D}) + P(S_m/D)P(D)$$

where

\bar{D} = event that common cause failure *does not* occur

D = event that common cause failure *does* occur

$P(S_m/\bar{D})$ obtained from binomial distribution, as before

$$P(S_m/D) = \begin{cases} 1.0 & \text{for } m=0 \\ 0 & \text{for } m>0 \end{cases}$$

$$P(\bar{D}) = p_d ; P(D) = 1 - p_d$$

Distribution between failure event types

$$p = p_i p_d = (1 - f_i) (1 - f_d) = 1 - f$$

Let

$$k_d = f_d/f = \frac{1 - p_d}{1 - p_i p_d} ; 0 \leq k_d \leq 1$$

or

$$p_d = \frac{1 - k_d}{1 - k_d p_i} = 1 - k_d (1 - p)$$

Special case:

$$p_i = p_d = \sqrt{p}$$

$$k_d = (1 - \sqrt{p})/(1 - p)$$

p	$p_i = p_d$	k_d
0.5	0.7071	0.5858
0.6	0.7746	0.5635
0.7	0.8367	0.5445
0.8	0.8944	0.5279
0.9	0.9487	0.5132

Parametric Results For $m = 6$, $p_i = 0.94$, $p_c = 0.98$, $P = 0.8$

k_d	n	p_i	p_d	$\frac{p_c}{p_c^*}$ (using $\frac{p_i + p_d}{2}$ for cost)
0.2	6	0.9777	0.9945	2.861
	7	0.9153	0.9793	2.368
	8	0.8533	0.9646	2.314
	9	0.7984	0.9520	2.355
	10	0.7506	0.9413	2.433
0.5	6	0.9800	0.9804	2.650
	7	0.9336	0.9377	2.279
	8	0.8965	0.9062	2.281
	9	0.8716	0.8862	2.368
	10	0.8576	0.8754	2.494
<u>$p_i = p_d$</u>				
0.5051	6	0.9800	0.9800	2.645
0.5167	7	0.9350	0.9350	2.274
0.5259	8	0.9014	0.9014	2.281
0.5314	9	0.8817	0.8817	2.377
0.5340	10	0.8727	0.8727	2.515
0.8	6	0.9858	0.9463	2.368
	7	0.9678	0.8859	2.224
	8	0.9628	0.8705	2.352
	9	0.9622	0.8687	2.526
	10	0.9621	0.8684	2.705
0.9	6	0.9904	0.9205	2.243
	7	0.9838	0.8728	2.233
	8	0.9832	0.8687	2.404
	9	0.9832	0.8687	2.589
	10	0.9832	0.8687	2.774

Session B, Submittal No. 6

Manuel I. Cruz
TRW

EARTH PENETRATION CONCEPTS
AND
MARS PENETRATOR ENTRY TRADES

MANNY CRUZ
TRW

Earth Penetration Concepts

TABLE 1

EARTH PENETRATOR RE-ENTRY TO IMPACT CONCEPTS

Concept	Control Mechanism	Ballistic Coefficient (kg/m ²)	Staging Conditions			Impact Conditions			Comments
			V (mps)	Gamma (degree)	h (km)	V (mps)	Gamma (degree)	1 σ Impact Error (meters/driver)	
Two Stage Ballistic Entry and High Altitude Drag Modulation	Drag modulation and staging from low B to high B	10 \leq B \leq 15000	180 to 250	90	30	300 to 900	90	~300/driven by low altitude winds \leq 9 mps	Very good terminal velocity and flight path control; large CEP considerations
Two Stage Ballistic Entry	Staging from low B to high B	10 \leq B \leq 15000	180 to 250	90	30	300 to 900	90	~600 to 900/high altitude density variation \leq 20 L/D low altitude winds \leq 9 mps	Very good terminal velocity and flight path control; large CEP considerations
Two Stage Ballistic Entry	Staging from moderate B to high B	730 \leq B \leq 15000	300 to 900	50	18	450 to 900	50	~300 to 600/high altitude density variation \leq 20 L/D low altitude winds \leq 9 mps	Plagued by lateral load carrying capability of impact system medium strength rock
Ballistic Entry Moderate B	None	2500	N/A	N/A	N/A	215	60	~300	Plagued by rebound and/or ricochet
Ballistic Entry High B	None	9800 \leq B \leq 15000	N/A	N/A	N/A	2100 to 3400	30	Driven by delivery system	Exceeds load carrying capability of impact system
Maneuvering Re-Entry Pitch/Bank	Lift Modulation	4900 \leq B \leq 15000	N/A	N/A	N/A	300 to 900	90	Driven by delivery system	Very good velocity and flight path control throughout impact error exceeds requirements

PENETRATION PREDICTION

$$D = 0.0031 (S) (N) \sqrt{W/A} (V-100), \quad V \geq 200 \text{ FT/SEC}$$

D = PENETRATION DEPTH (FT)

S = SOIL CONSTANT

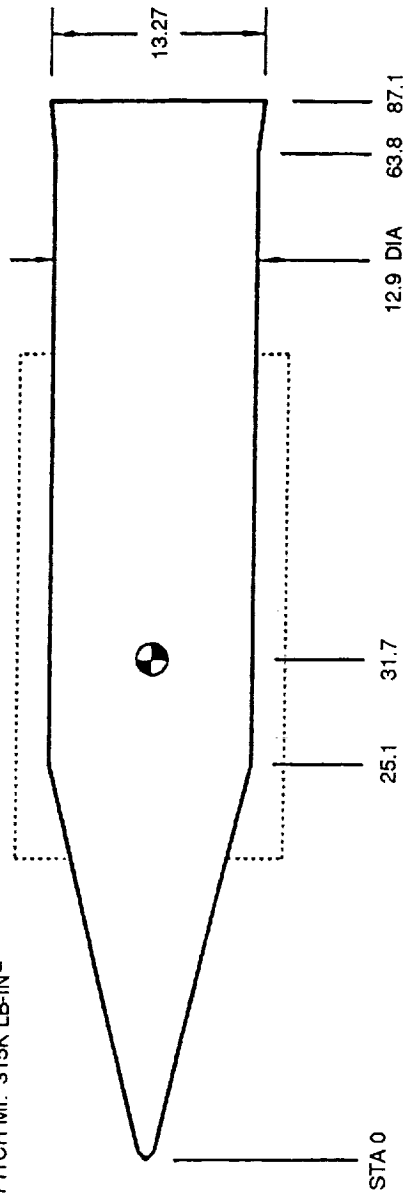
N = NOSE PERFORMANCE COEFFICIENT

W = WEIGHT (LB)

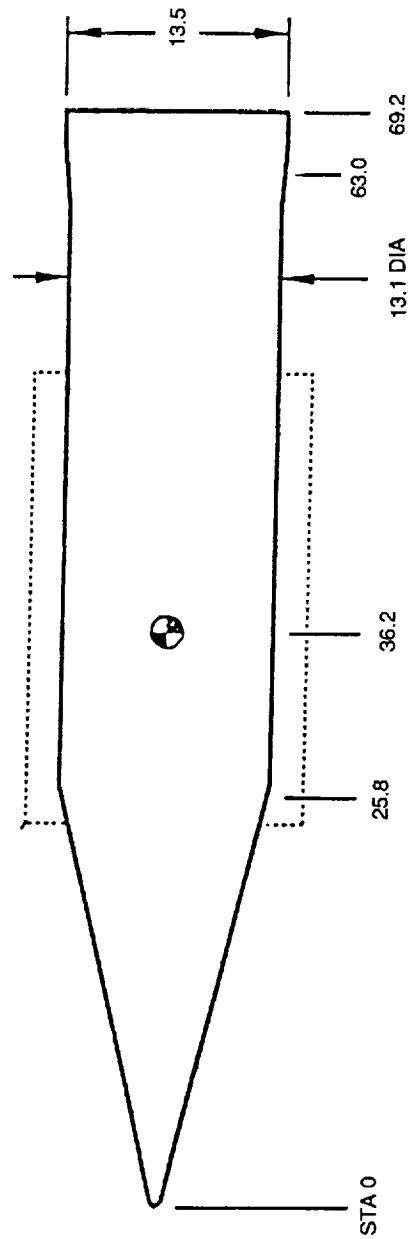
A = CROSS-SECTIONAL AREA (FT²)

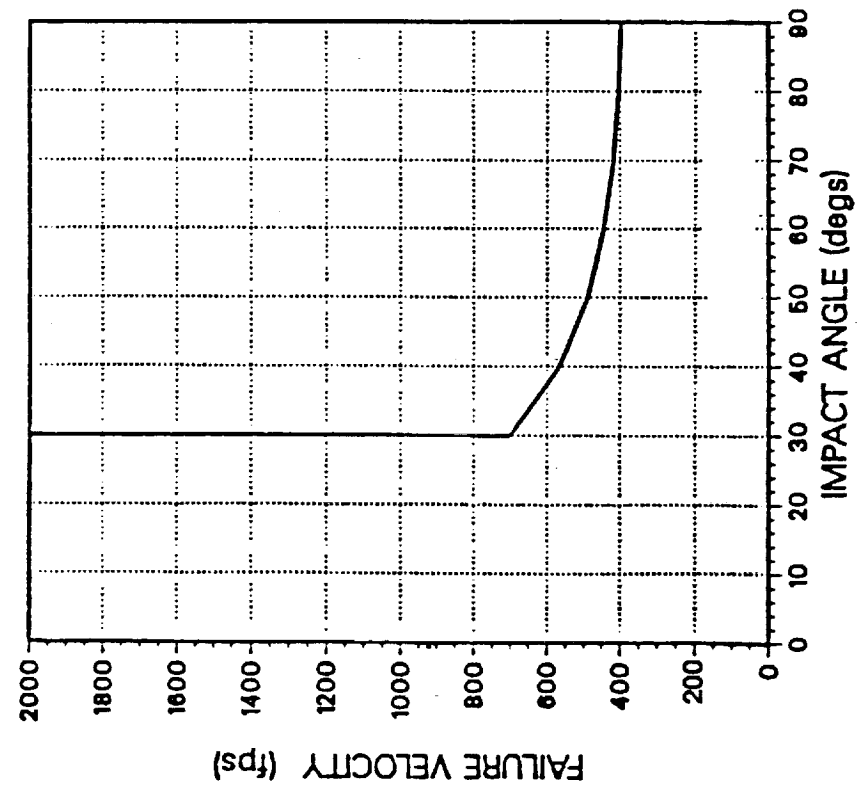
V = VELOCITY AT IMPACT (FT/SEC)

WEIGHT: 860 LBS
 ROLL MI: 14.5K LB-IN²
 PITCH MI: 315K LB-IN²

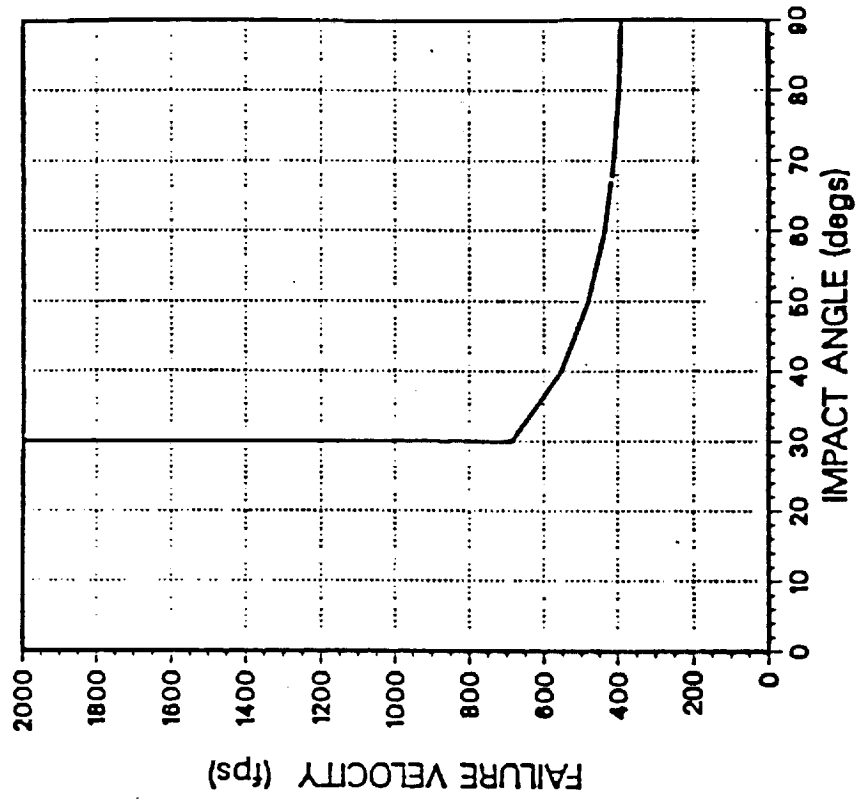


WEIGHT: 850 LBS (MAX)
 ROLL MI: 14.0K LB-IN²
 PITCH MI: 265K LB-IN²

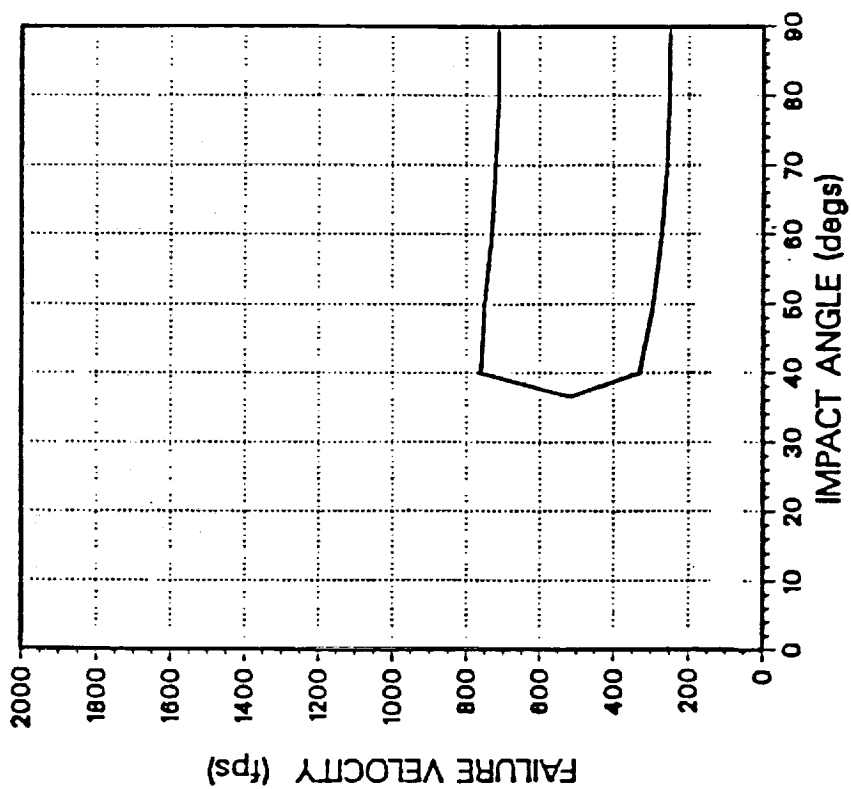




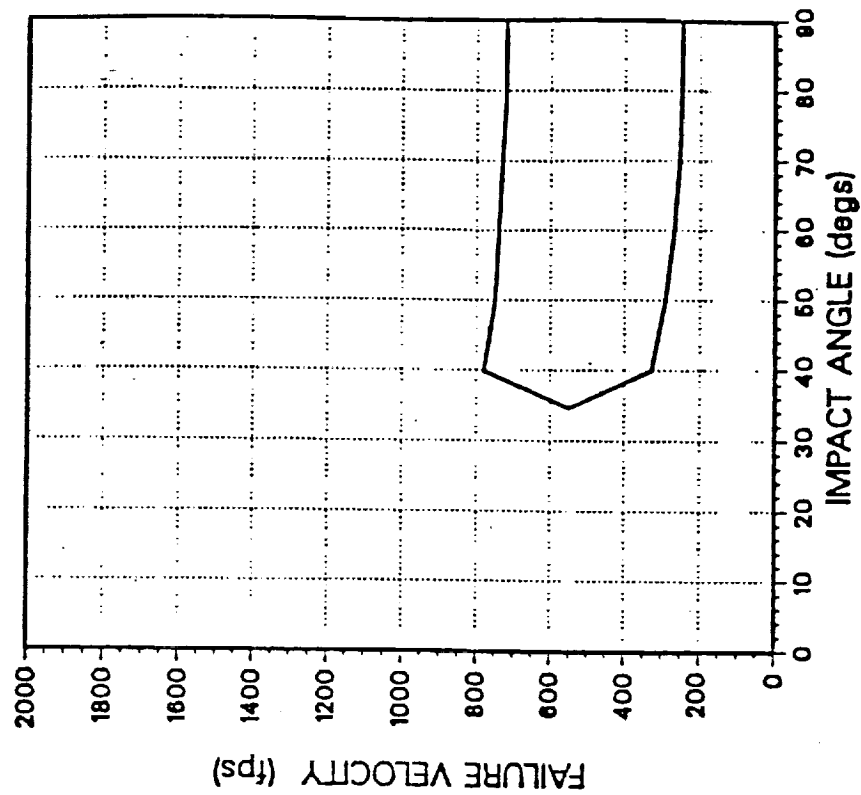
Baseline Design - Target Type 3 (Solid)
-2.0 Degrees Angle of Attack



860 lb Design Excursion 1 - Target Type 3 (Solid)
-2.0 Degrees Angle of Attack

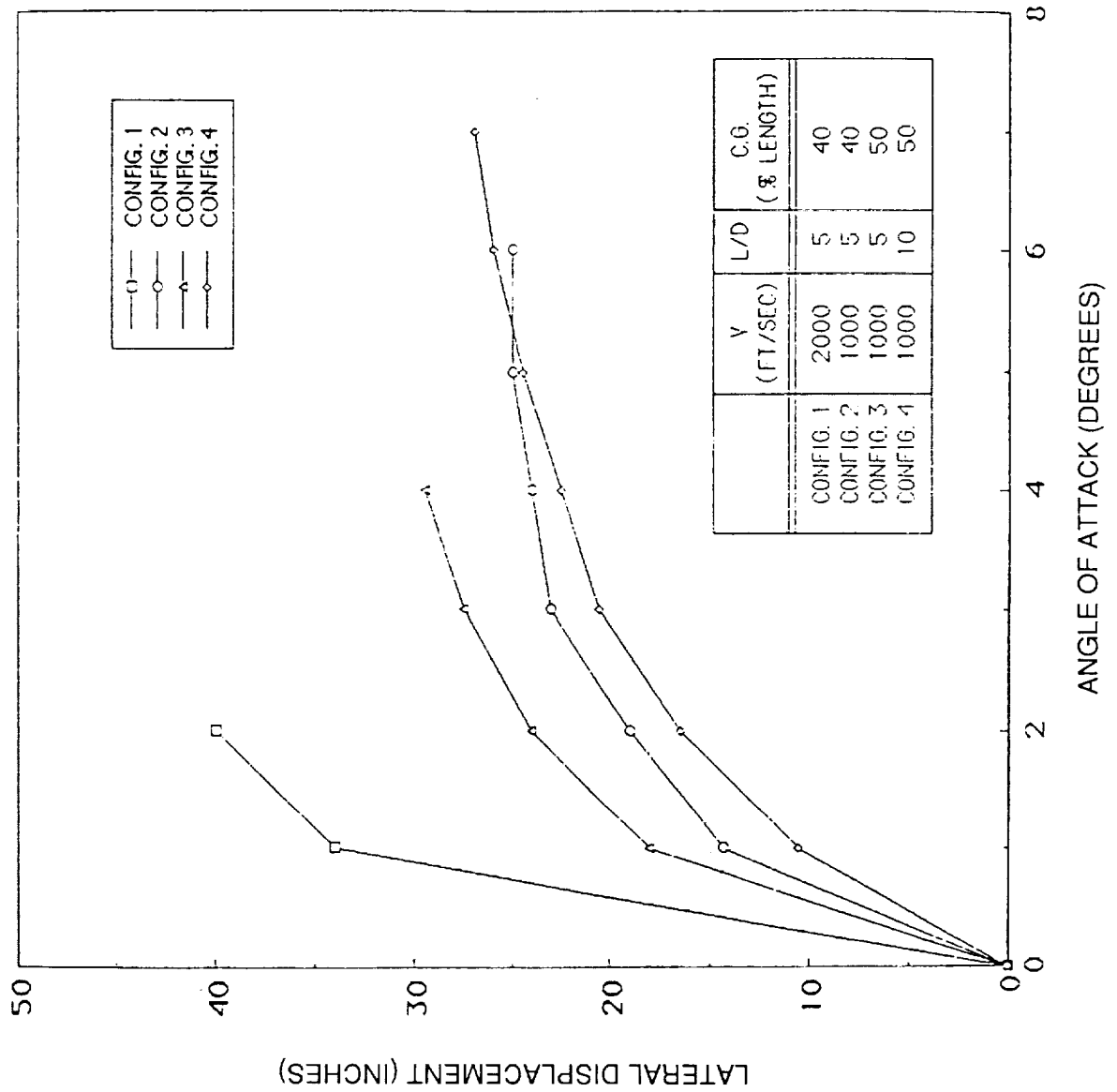


Baseline Design - Target Type 4 (Soil/Rock)
-2.0 Degrees Angle of Attack

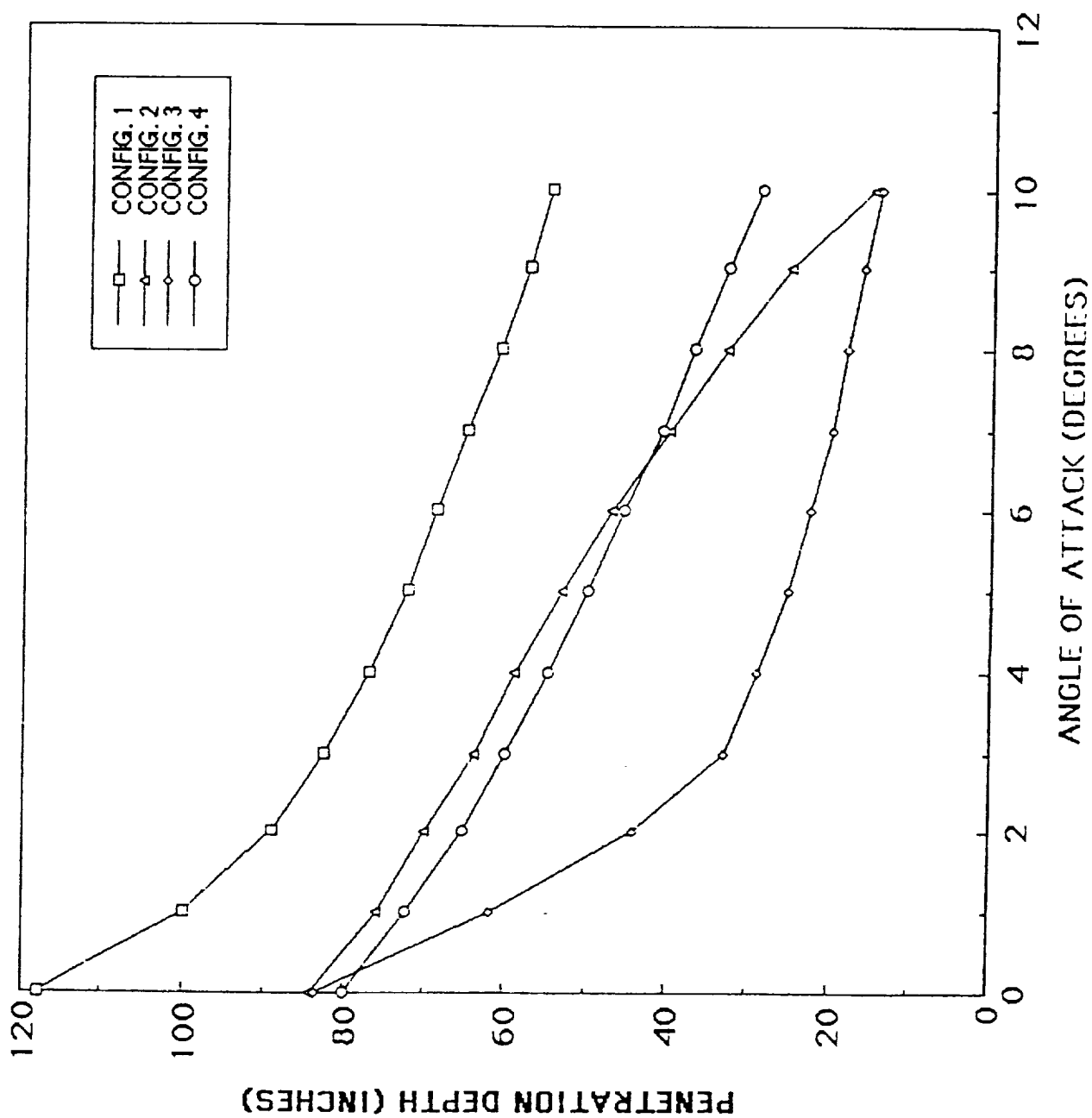


860 lb Design Excursion 1 - Target Type 4 (Soil/Rock)
-2.0 Degrees Angle of Attack

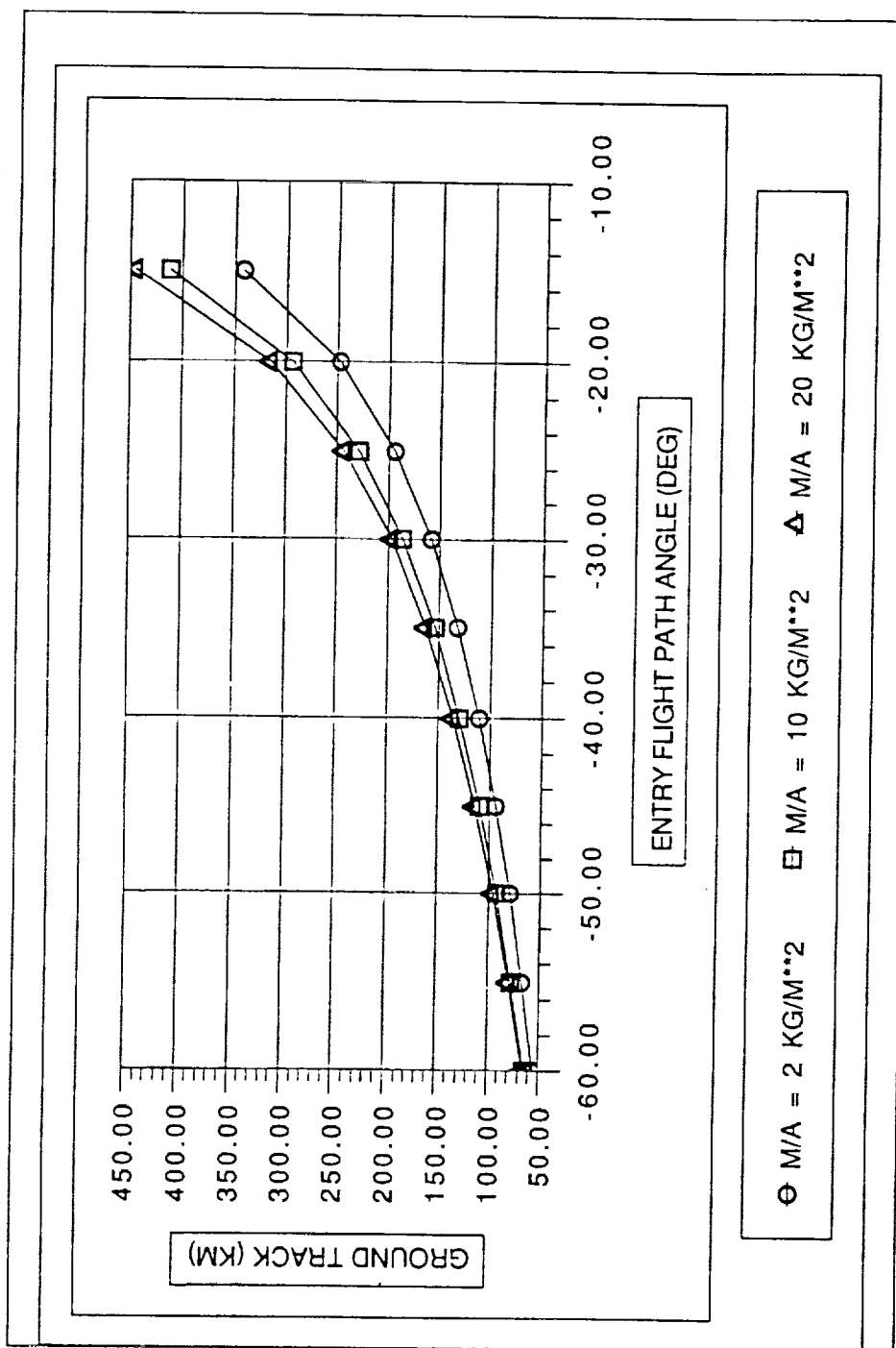
LATERAL DISPLACEMENT VS ANGLE OF ATTACK FOR 4 PENETRATOR CONFIGURATIONS
(MEDIA - SAND)

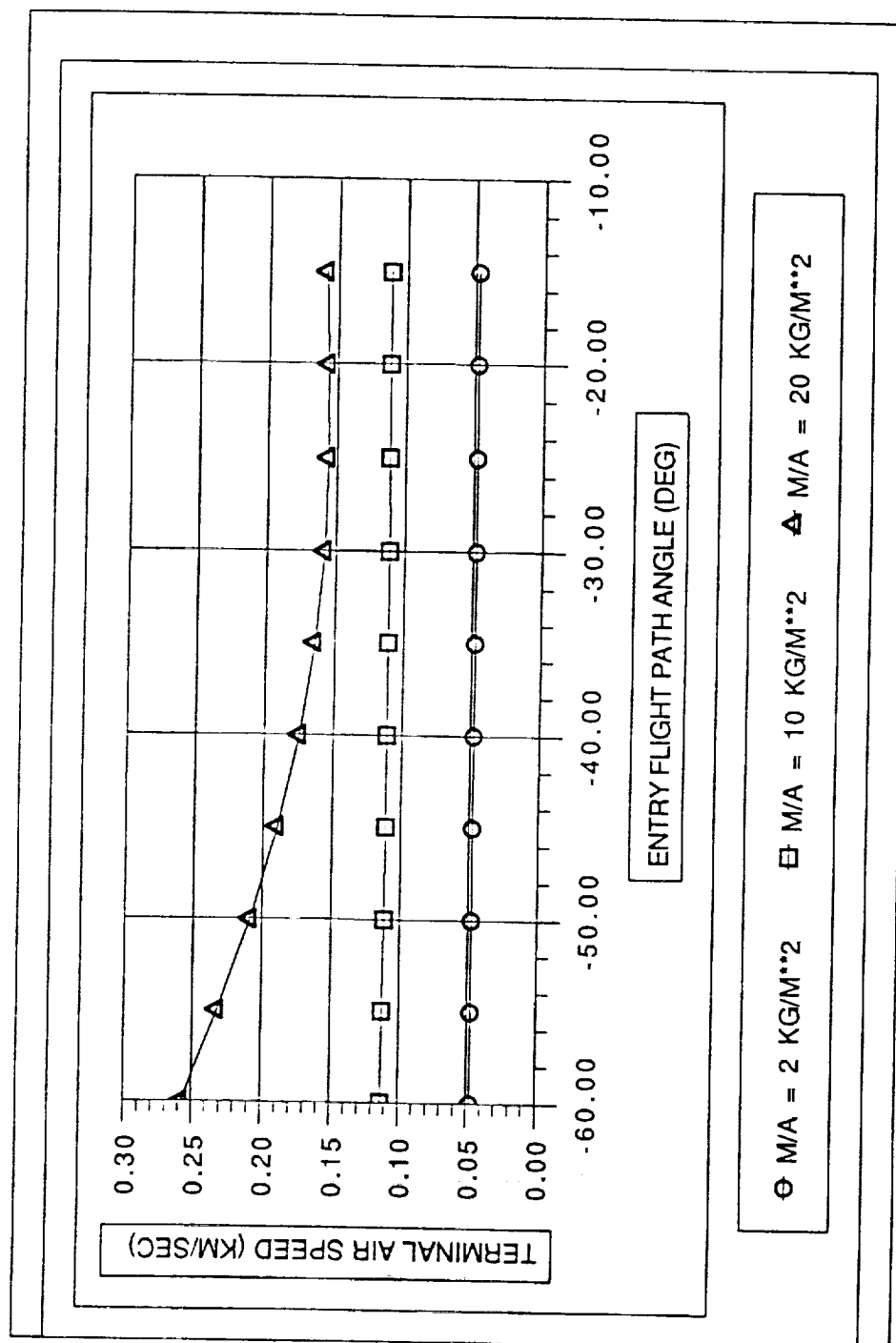


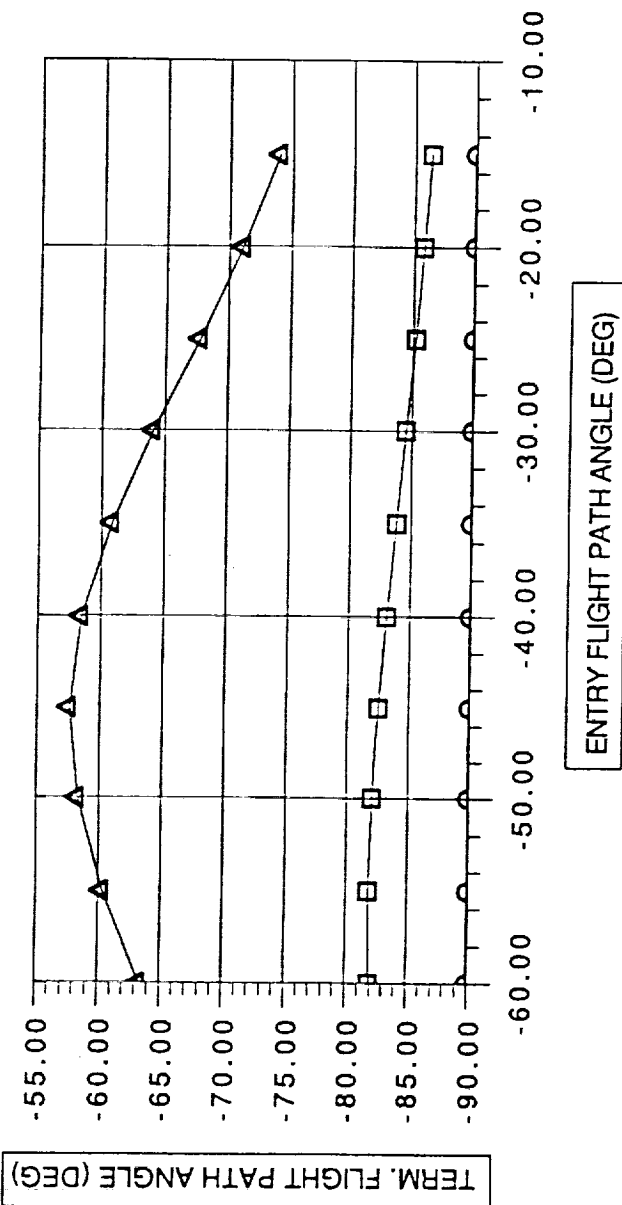
PENETRATION VS. ANGLE OF ATTACK FOR 4 PENETRATOR CONFIGURATIONS (MEDIA - SAND)



Mars Penetrator Entry Trades

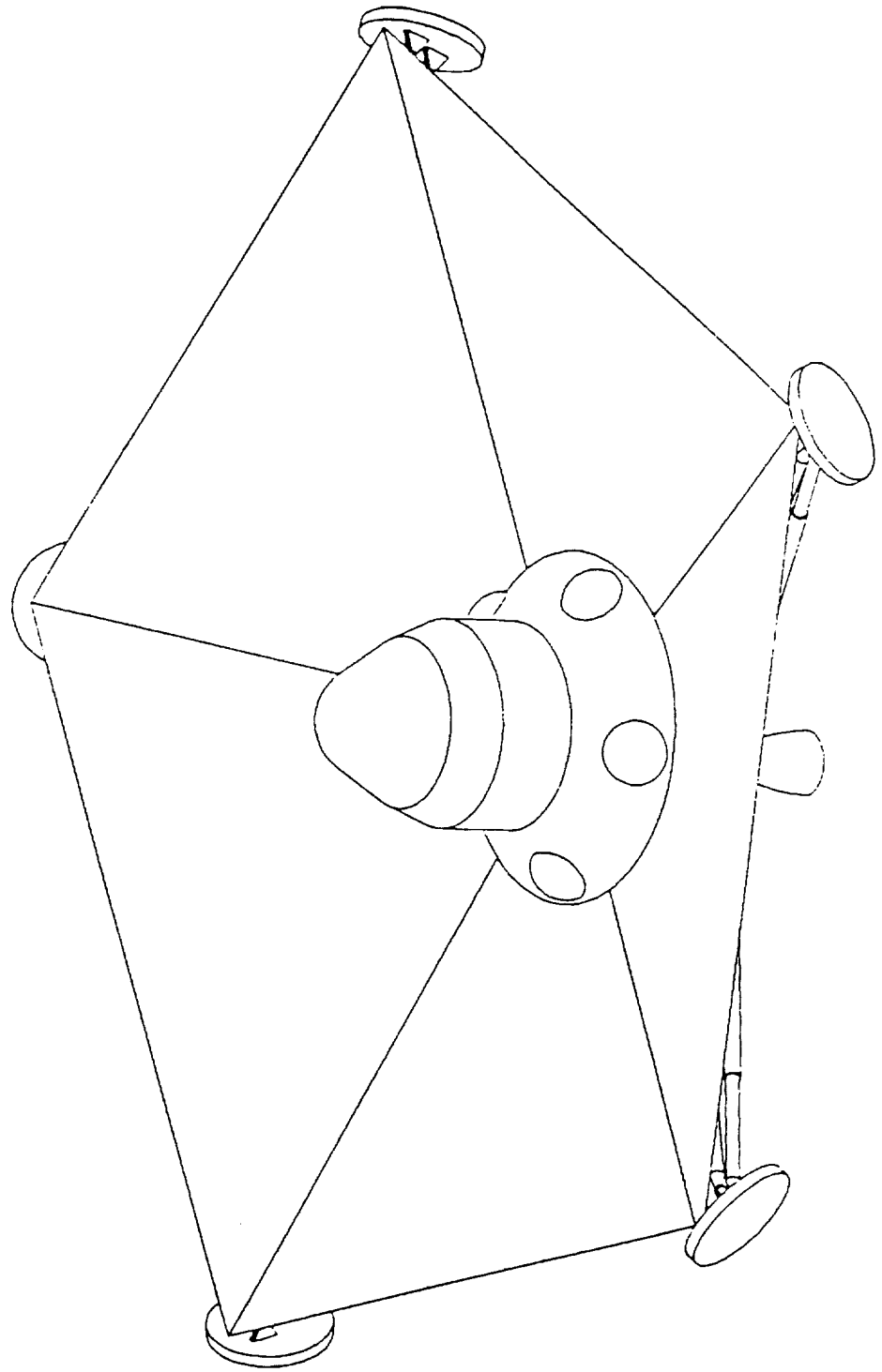






\ominus M/A = 2 KG/M**2 \boxminus M/A = 10 KG/M**2 \blacktriangle M/A = 20 KG/M**2

AEROBRAKE DESIGN

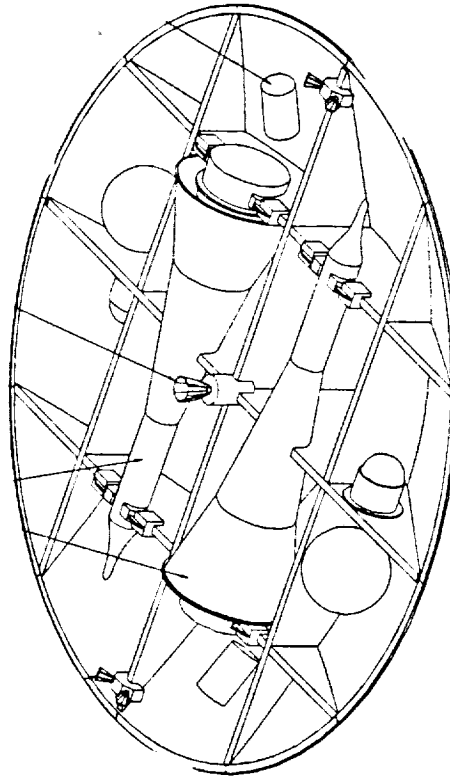


Session B, Submittal No. 7

Richard P. Reinert
Ball Space Systems Division



CRAF PENETRATOR-LANDER FOR MARS



PREPARED FOR MGNM WORKSHOP AT JPL

02/06/90

W. Boynton
University of Arizona

R. Reinert
Ball Space Systems Division



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BRIEFING OUTLINE

PRELIMINARY MISSION REQUIREMENTS

POLAR PENETRATOR DERIVED FROM CRAF CONFIGURATION

PV FUNCTIONAL SHOWS CENTRAL ROLE OF C&DH

SCIENCE OBJECTIVES SIMILAR TO CRAF

BASIC INSTRUMENT COMPLEMENT IS IDENTICAL EXCEPT FOR GRS

POLAR PENETRATOR MASS PROPERTIES MATCH BASELINE

POLAR PENETRATOR CONFIGURATION COMPATIBLE WITH BASELINE AEROSHELL

POLAR PENETRATORS OCCUPY ONE AEROSHELL

MPP AEROSHELL/PENETRATOR SYSTEM

PENETRATOR EXTRACTION FORM AEROSHELL

PENETRATOR DEPLOYMENT



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Systems
Division

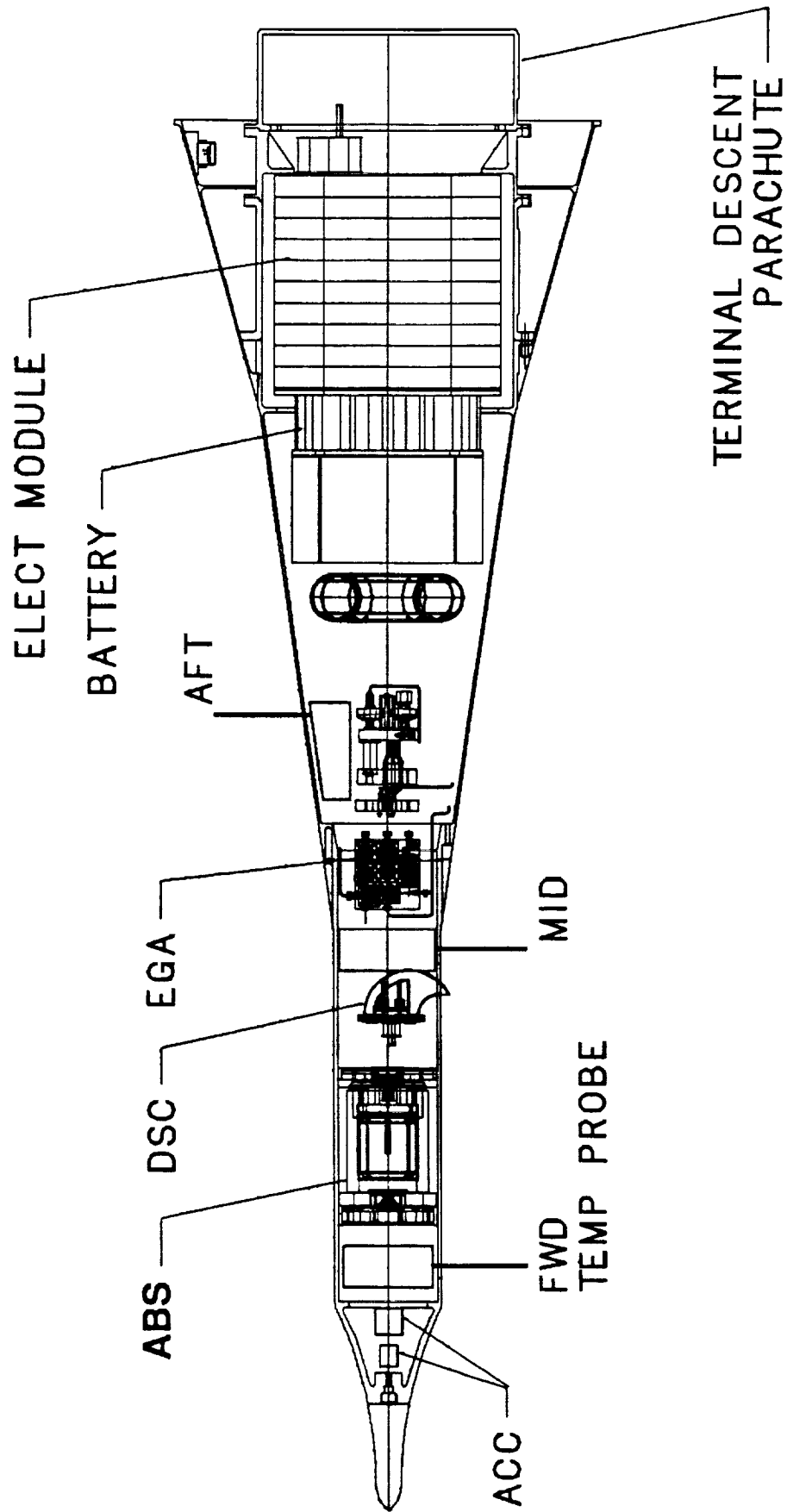
PRELIMINARY MISSION REQUIREMENTS

REQUIREMENT	MPP	CRAF
SCIENCE PAYLOAD	MPP INSTRUMENT COMPLEMENT AS DESCRIBED BELOW	CRAF INSTRUMENT COMPLEMENT
MISSION DURATION: (APPROX)	CRUISE 330 D MARS ORBIT 30-60 D OPERATIONAL 7-14 D	6 YR 1 YR (NUCLEUS) 6-7 D
LAUNCH DATE	1998	1995
LAUNCH/TRANSPORT IV/CENTAUR/	TITAN IV/CENTAUR/ MODIFIED M/O BUS	TITAN MM/II BUS
DEPLOYMENT	DEORBIT/MARS ATMOSPHERE ENTRY DUAL MANIFEST 120 DEG AEROSHELL 30 KG/M**2 M = 2 PARACHUTE EXTRACTION PARACHUTE PROVIDES IMPACT VELOCITY	FREE-FLIGHT TO NUCLEUS ONBOARD PRO- PULSION MODULE PROVIDES IMPACT VELOCITY
IMPACT CONDITIONS S-RANGE VELOCITY MIN DEPTH	3 - 50 50 M/S 1 M	3-200 20-50 M/S 1 M
COMM RELAY BITS/DAY (AVG) BER RANGE (MAX) PASSES/DAY PASS DURATION	256 K 1X10-5 600 KM 2 M 5 M MIN	256 K 1X10-5 100 KM 2 M 10 SEC MIN

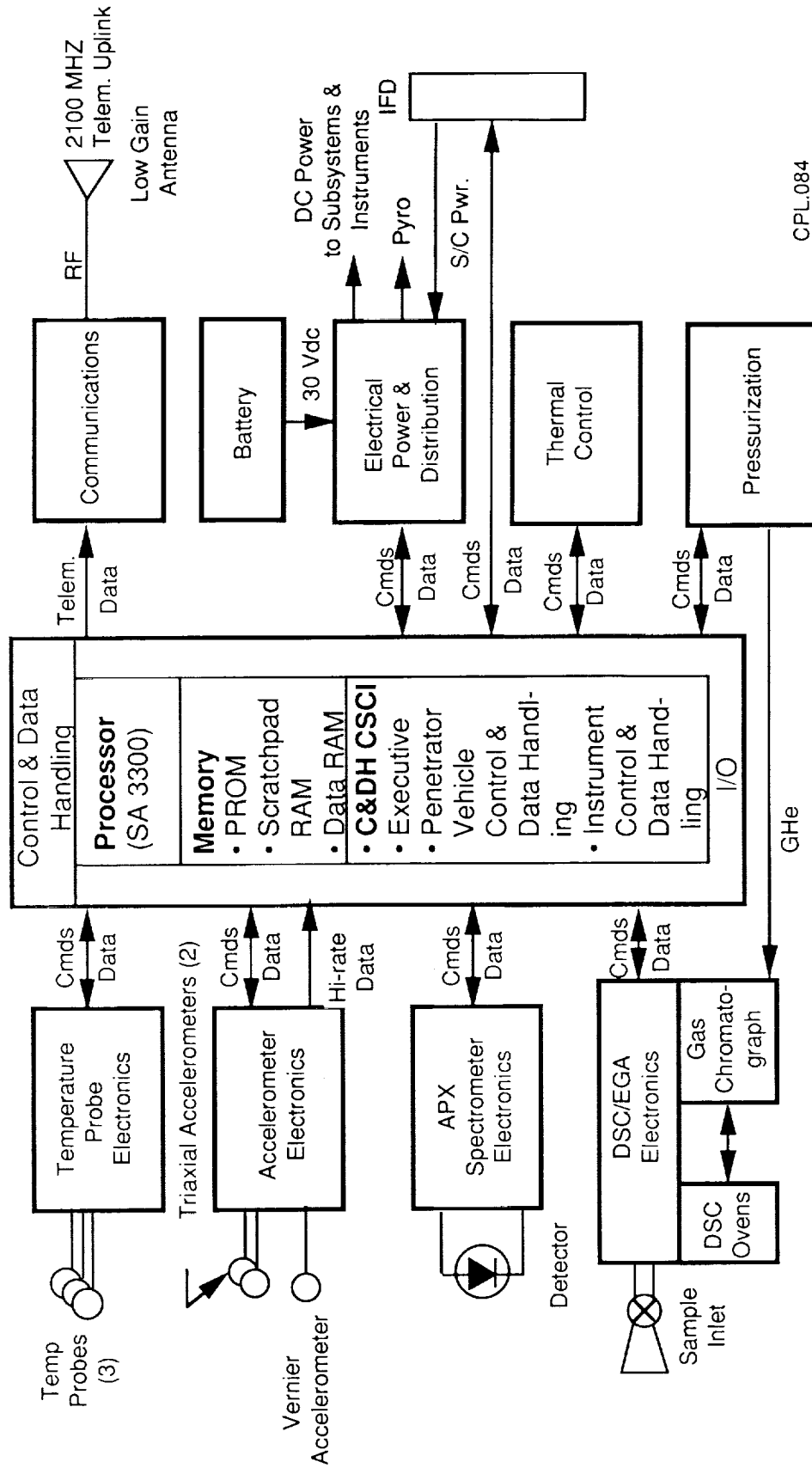


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POLAR PENETRATORS DERIVED FROM CRAF CONFIGURATION



MPP FUNCTIONAL ARRANGEMENT IS SIMPLER



- Approx 90% commonality with CRAF PENL



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SCIENCE OBJECTIVES SIMILAR TO CRAF

- Measure strength, depth of penetration
- Detect stratigraphic layers
- Measure initial temperature profile
- Measure thermal diffusivity
- Determine elemental abundances of most major and minor and a few trace elements
- Determine the temperature and enthalpy of phase changes as a diagnostic of the mineralogical/molecular form of the major components
- Determine the identity and vapor pressure of major and trace volatile compounds as a function of temperature



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BASIC INSTRUMENT COMPLEMENT IS IDENTICAL EXCEPT FOR GRS

ELEMENT	BASIC FUNCTION
SEVEN ACCELEROMETERS (ACCs)	MEASURE THE DECELERATION PROFILE ON IMPACT
THREE THERMAL PROBES (TPs)	MEASURE TEMPERATURES AND THERMAL DIFFUSIVITY AS A FUNCTION OF DEPTH
ALPHA BACKSCATTER SPECTROMETER (ABS)	MEASURE IN-SITU ELEMENTAL COMPOSITION
DIFFERENTIAL SCANNING CALORIMETER (DSC)	DETERMINE THE PHASE COMPOSITION OF THE ICES AND MINERALS PRESENT
EVOLVED GAS ANALYZER (EGA)	DETERMINE THE MOLECULAR COMPOSITION OF ENTRAPPED GASES EVOLVED FROM DSC HEATING



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POLAR PENETRATOR MASS PROPERTIES MATCH BASELINE

ITEM	MASS, KG (1)		RATIONALE
	CRAF	MPP	
MASS SPEC	3.7	0.5	REPLACE GRS WITH ABS
TEMP PROBE GROUP	1.5	1.5	IDENTICAL
ACCELEROMETER GROUP	0.5	0.5	"
DSC ANALYZER	1.9	1.9	"
EGA ANALYZER	5.0	5.0	"
INSTRUMENT ELECTRONICS	2.8	2.3	DELETE GRS HVPS
STRUCTURE	19.5	19.5	IDENTICAL
ELECTRICAL POWER	9.2	9.3	648 W-HR
C&DH	2.1	2.1	IDENTICAL
COMMUNICATIONS	0.6	0.6	16 KBPS/100 KM
THERMAL CONTROL	0.6	0.6	IDENTICAL
PROPULSION	6.9	-	NOT NEEDED
TOTAL	54.3	44.2	

NOTES:

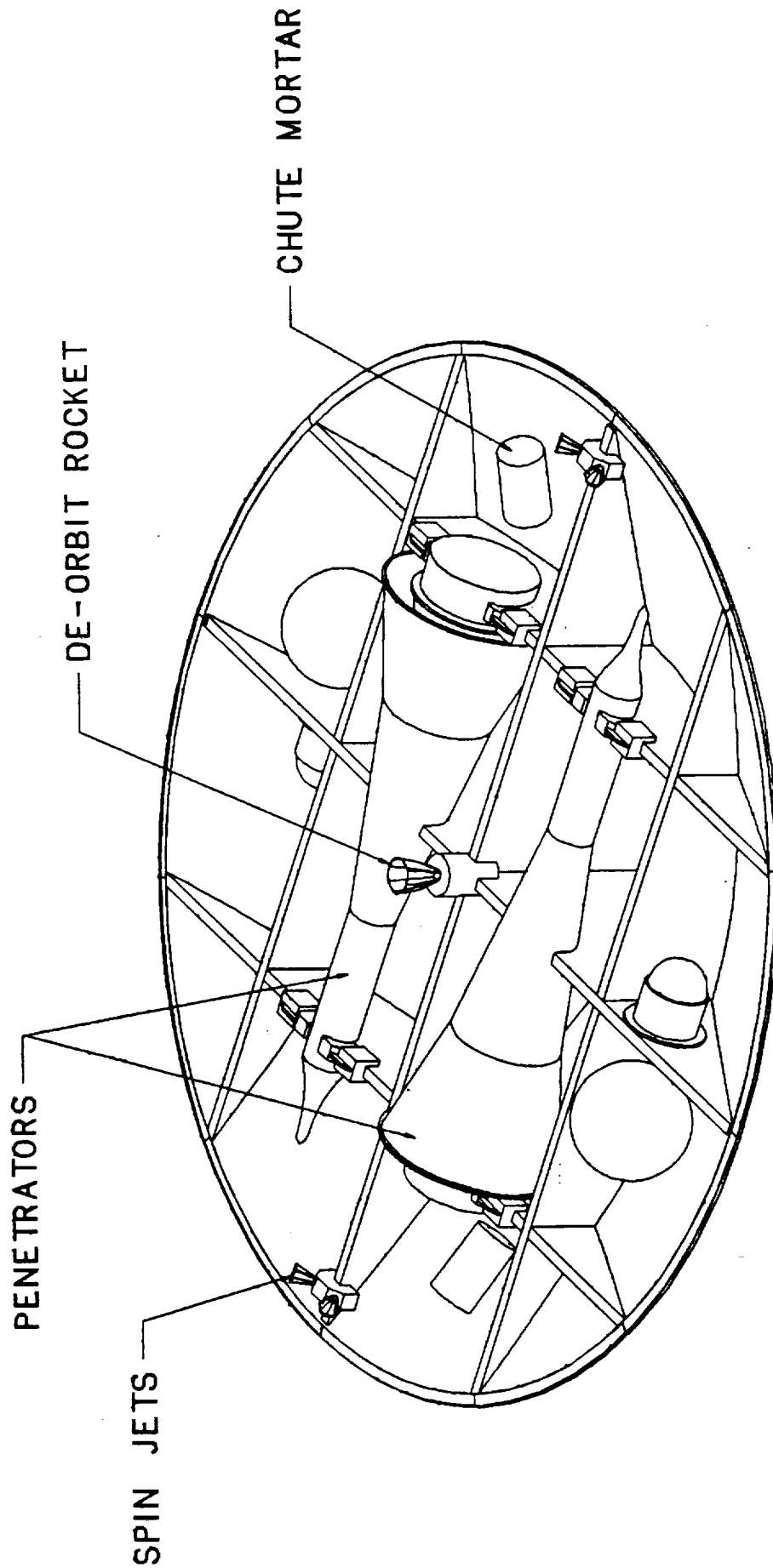
(1) CONTINGENCY INCLUDED AT SUBSYSTEM LEVEL

(2) BASELINE MASS = 46.3 KG



Space
Systems
Division

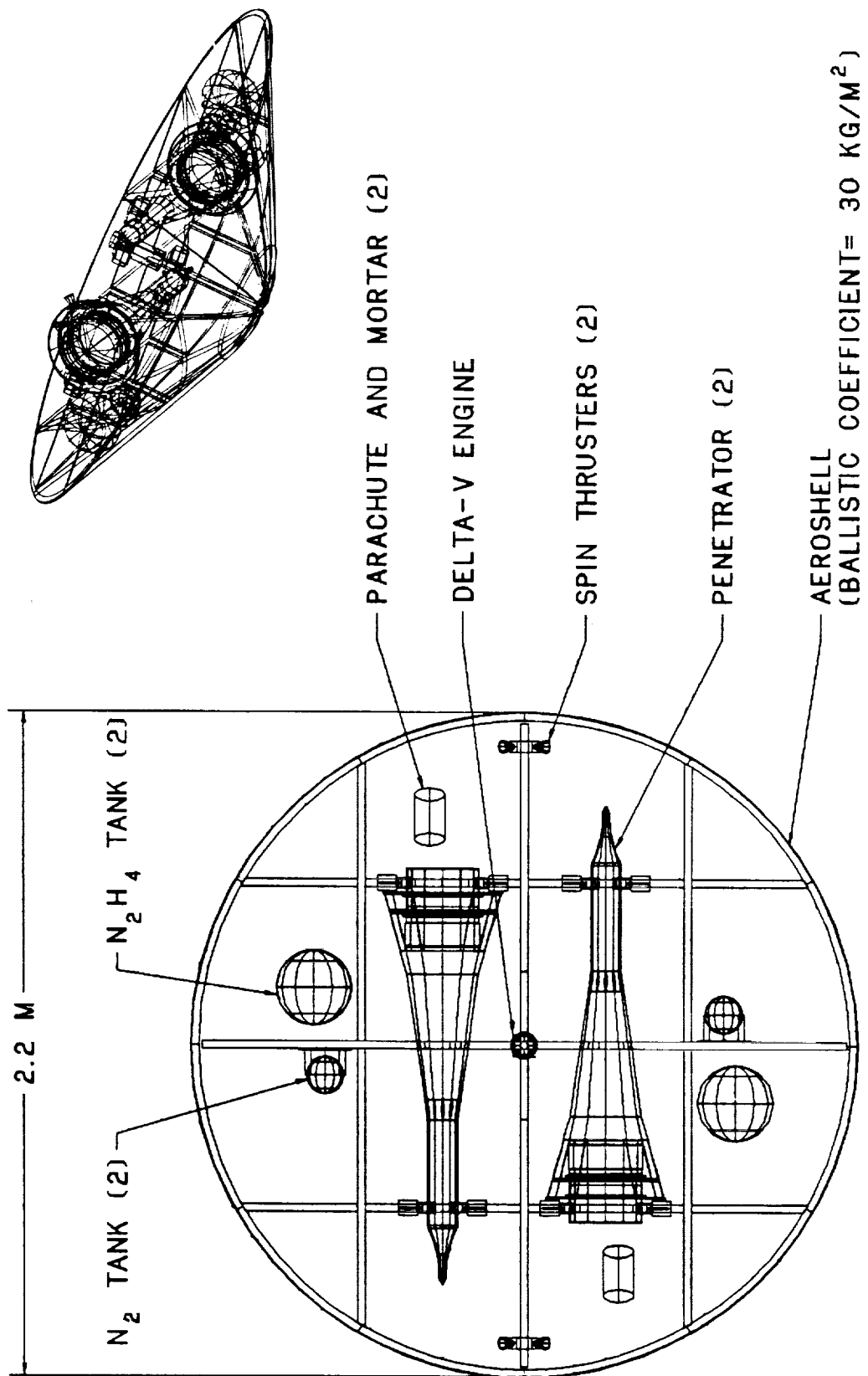
POLAR PENETRATOR CONFIGURATION COMPATIBLE WITH BASELINE AEROSHELL





Space
Systems
Division

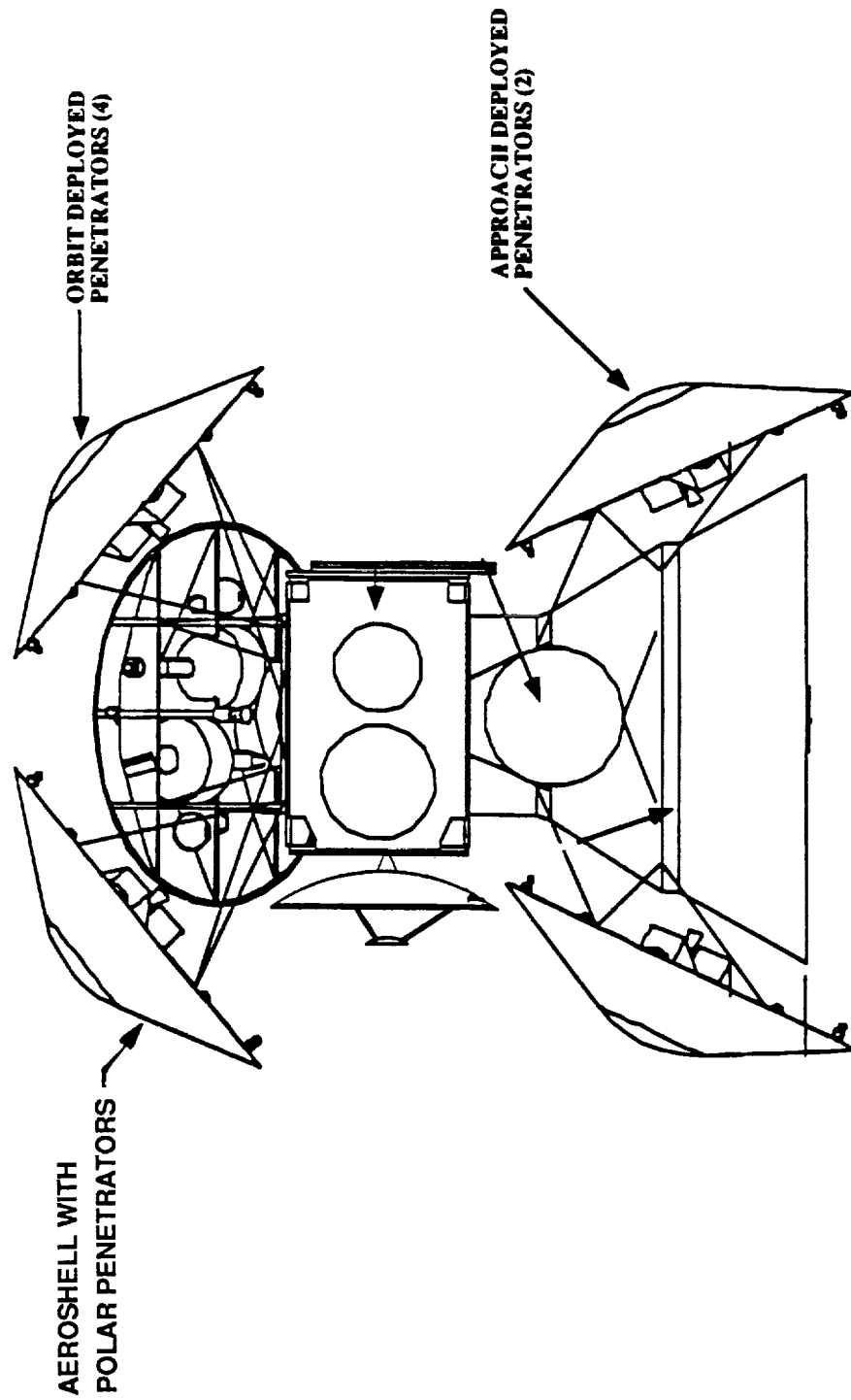
MPP AEROSHELL/PENETRATOR SYSTEM





Space
Systems
Division

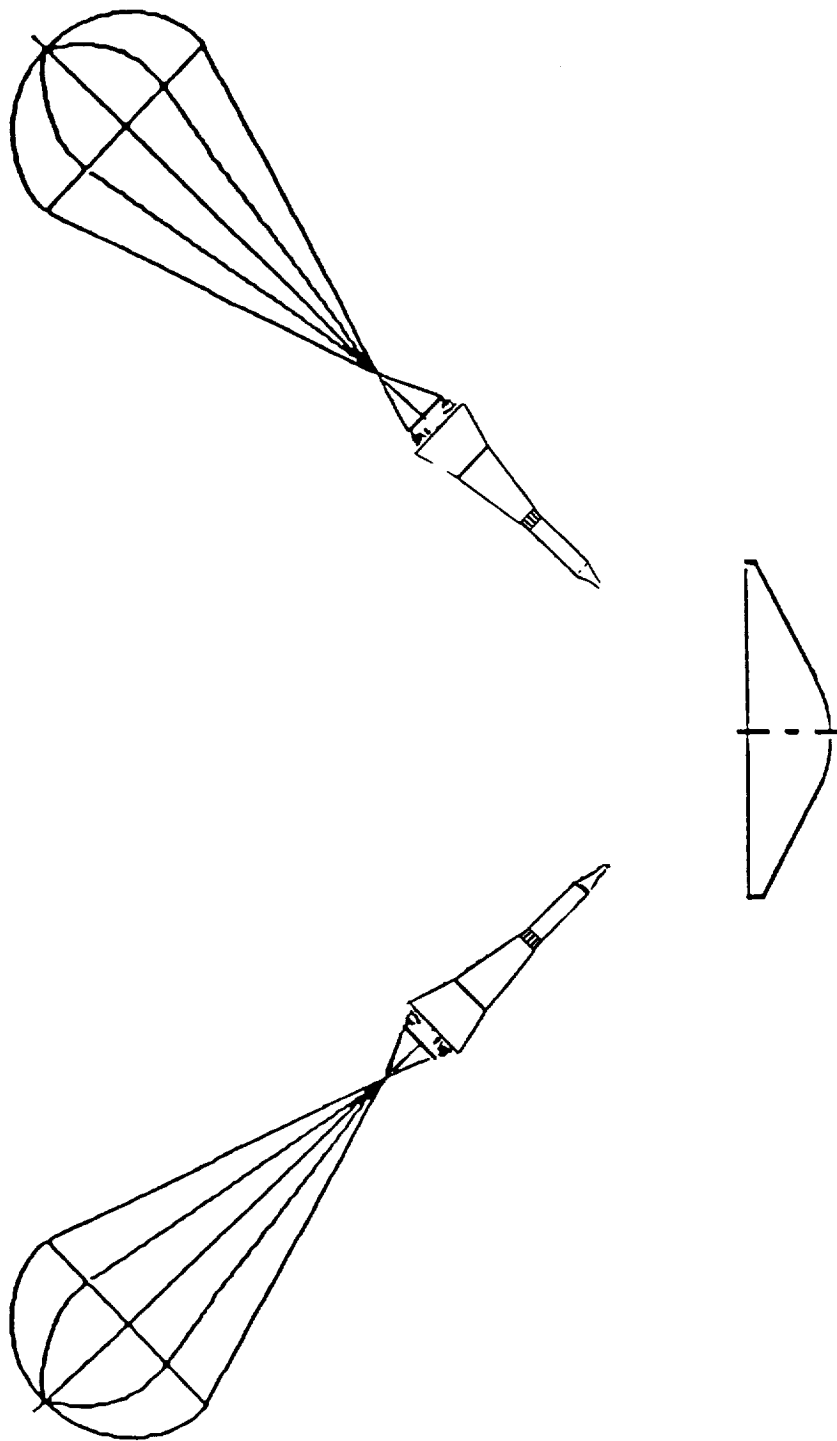
POLAR PENETRATORS OCCUPY ONE AEROSHELL





Space
Systems
Division

PENETRATOR EXTRACTION FROM AEROSHELL



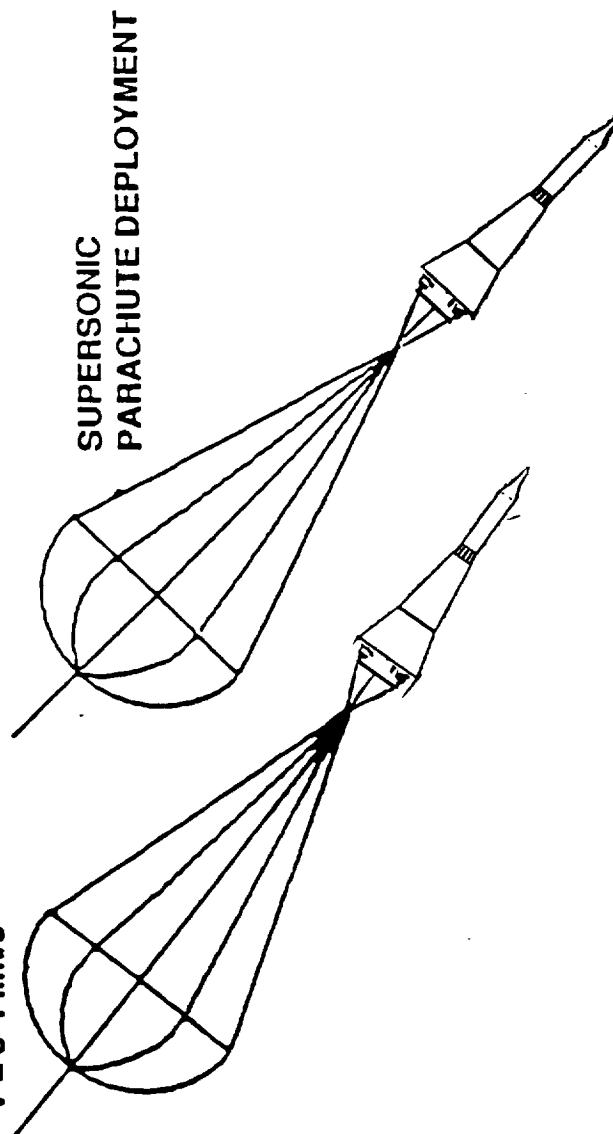
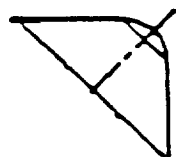


Space
Systems
Division

PENETRATOR DEPLOYMENT

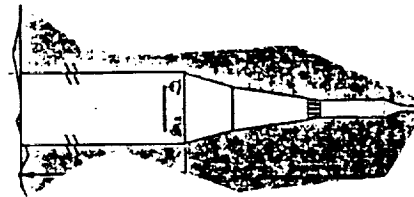
ENTRY

$V = 3-4 \text{ km/s}$



SUPERSONIC
PARACHUTE DEPLOYMENT

SURFACE IMPACT
 $V = 60-100 \text{ m/sec}$



IMPACT

- ANGLE OF ATTACK – WIND EFFECT
- ANGLE OF INCIDENCE – SURFACE UNDULATIONS
- SURFACE HARDNESS – PENETRABILITY INDEX

OPEN-CONCERN
ACCEPTABLE UP TO 30°
3 – 200

COMMUNICATIONS

- ACCESSIBILITY TIME
- BIT RATE, LINK BUDGET

ASSUME 2 TO 4 MINUTES PER PASS
12.5 Kbit/sec

Session B, Submittal No. 8

Joe D. Gamble
Johnson Space Center

MARS GLOBAL NETWORK MISSION WORKSHOP

ENTRY SYSTEM DESIGN CONSIDERATIONS

J. Gamble - NASA/Johnson Space Center

Introduction

This section addresses some of the design issues concerned with the specific workshop question, "What is the best entry system - fixed or deployed aeroshells; parachutes or direct impact?" To address these questions some information about the entry conditions in the Mars environment is required. Results from the 90 day human exploration initiative study were used as a reference point. The MRSR pre-phase A study results were also considered. Finally some parametric data was generated to specifically address the GNM entry design question.

Reference Mission

The 90 day study considered two flight systems each consisting of an orbiter/carrier vehicle with six aeroshells as shown in Figure 1. Each aeroshell contains two penetrator landers as shown in Figure 2 that use parachutes to extract them from the aeroshell just prior to landing. The rigid aeroshells are deployed from the carrier vehicle and spin stabilized at 60 rpm. Small propulsion systems provide the delta V required for the desired atmospheric entry conditions. The aeroshells do not have an active guidance and control system.

The aeroshell design incorporates a rigid conical aeroshell with a spherical nose cap. The aeroshell diameter is 2.2 m and has an entry mass of approximately 110 kg, yielding a ballistic coefficient of 30 kg/m². The aeroshell uses an ablative heat shield.

Two of the six aeroshells are deployed 2-10 days prior to Mars arrival in order to land at polar sites. The other four aeroshells are deployed after capture into a 1/5 Sol Mars orbit.

Mars Approach Deployed Aeroshells

One of the primary concerns in the MRSR study was the ability to achieve the proper entry conditions during the Mars approach. The entry corridor is bounded by the skip out and maximum allowable g load boundaries as shown in figure 3. Figure 4 shows the entry corridor limits versus L/D for an entry velocity of 6 km/sec. The total corridor width is summarized in figure 5 and shows that the corridor is nearly independent of the ballistic coefficient. The ballistic

coefficient determines whether the vehicle flies higher or lower in the atmosphere during the early portion of the entry. While the MRSR was concerned with aerocapture during the approach phase, the results are also applicable to the entry case. The estimated corridor width for ballistic vehicles are shown on figure 5. For a maximum entry load of 5 g's, the total corridor width is less than one degree. The corridor width increases to 3 and 5.5 for 10 and 15 g limits respectively. The MRSR study concluded that a minimum corridor width of approximately 3 degrees was required in order to accommodate navigation and atmosphere uncertainties. In order to achieve this accuracy, optical navigation was baselined for the study and trajectory corrections were considered within a few hours of entry.

GNM aeroshells deployed several days prior to entry and not having an active guidance and control system will almost certainly require much larger entry corridors than are necessary for the MRSR. It is very possible that a minimum corridor width of at least 5-10 degrees will be required. Figure 6 shows some preliminary results for the aeroshell defined by the 90 day study at an entry velocity of 6 km/sec at 125 km altitude. The figure indicates that g loads in excess of 20 g's will be required to provide a corridor width of 10 degrees. Figure 7 shows that for a 10 degree corridor width, downrange dispersions of +/- 2-5 degrees will occur for nominal entry angles of 15-20 degrees. These results were obtained from three degree-of-freedom simulations entering in a polar plane.

One proposal for decreasing the landing footprint dispersions is to enter at a much steeper entry angle. The results of entering at -35 and -45 degrees are shown as a function of ballistic coefficients in figure 8. The downrange dispersion for 10 degrees change in entry angle is less than one degree although it does increase as the ballistic coefficient increases. One of the primary problems with the steep entry angle is the large load factors that result. Figure 9 shows the maximum g loads (Earth g's) resulting from entry at -35 and -45 degrees. Load factors on the order of 40 - 60 g's result from these steep angles.

Deployable Aeroshell Considerations

Use of deployable aeroshell configurations will in general preclude the use of ablator heat shields and the ballistic coefficient will have to be small enough to limit the aeroheating during entry. To achieve a ballistic coefficient of 10 kg/m² using the 90 day study mass of 110 kg would require an aeroshell diameter of approximately 3.8 m while a diameter of 8.5 m would be required to achieve a ballistic coefficient of 2 kg/m². It would appear that use of deployed aeroshells of this size would have significant problems operating at 40-60 g's during entry. For this reason it is

questionable whether use of deployable aeroshells for entry during Mars approach is a viable concept.

Mars Orbit Deployed Aeroshells

The lower entry velocity for aeroshells deployed from Mars orbit present much less of a problem than for those deployed during approach. Figure 10 shows that entry corridors of 15 degrees are possible at less than 10 g maximum load. Because the navigation is much better defined for the orbit deployed aeroshell than for the approach deployed case, the entry angle dispersions will be much less. Figure 11 indicates that for entry angle dispersions of ± 1 degree, the dispersion in the downrange landing site will be well within ± 1 degree. Aerodynamic heating for the orbit entry cases will also be much lower than for the approach deployed aeroshells. It would appear that these advantages definitely outweigh the delta V penalty associated with capturing the aeroshells into Mars orbit.

Parachute Considerations

One of the primary concerns with use of parachutes for the final surface delivery of the instrument packages is whether acceptable deployment conditions can be achieved during the aeroshell entry. The Viking program used supersonic deployed parachutes which were required because of the uncertainty in the Mars atmosphere. In general deployment of parachutes up to around Mach 2 (approximately 500 m/sec at Mars) is considered well within the state of the art. Figure 12 shows the aeroshell velocity at 5 and 10 km altitude as a function of entry angle for the 30 kg/m² configuration with an entry velocity of 3.6 km/sec at 125 km altitude. The aeroshell is seen to be subsonic at both altitudes for the range of entry angles shown. Figure 13 shows the variation of the aeroshell velocity at 10 km altitude for various dispersions in the atmosphere. The low density cool COSPAR atmosphere results in barely supersonic conditions for the 30 kg/m² configuration and even a severe 50% decrease in atmospheric density only produces a Mach 2 case. Therefore use of parachutes for landing of the payload should not present any significant deployment problems.

The bibliography lists several references with some applications to the Mars entry problem. A number of these also have extensive bibliographies.

MSNM STOWED CONFIGURATION (ONE AEROSHELL REMOVED FOR CLARITY)

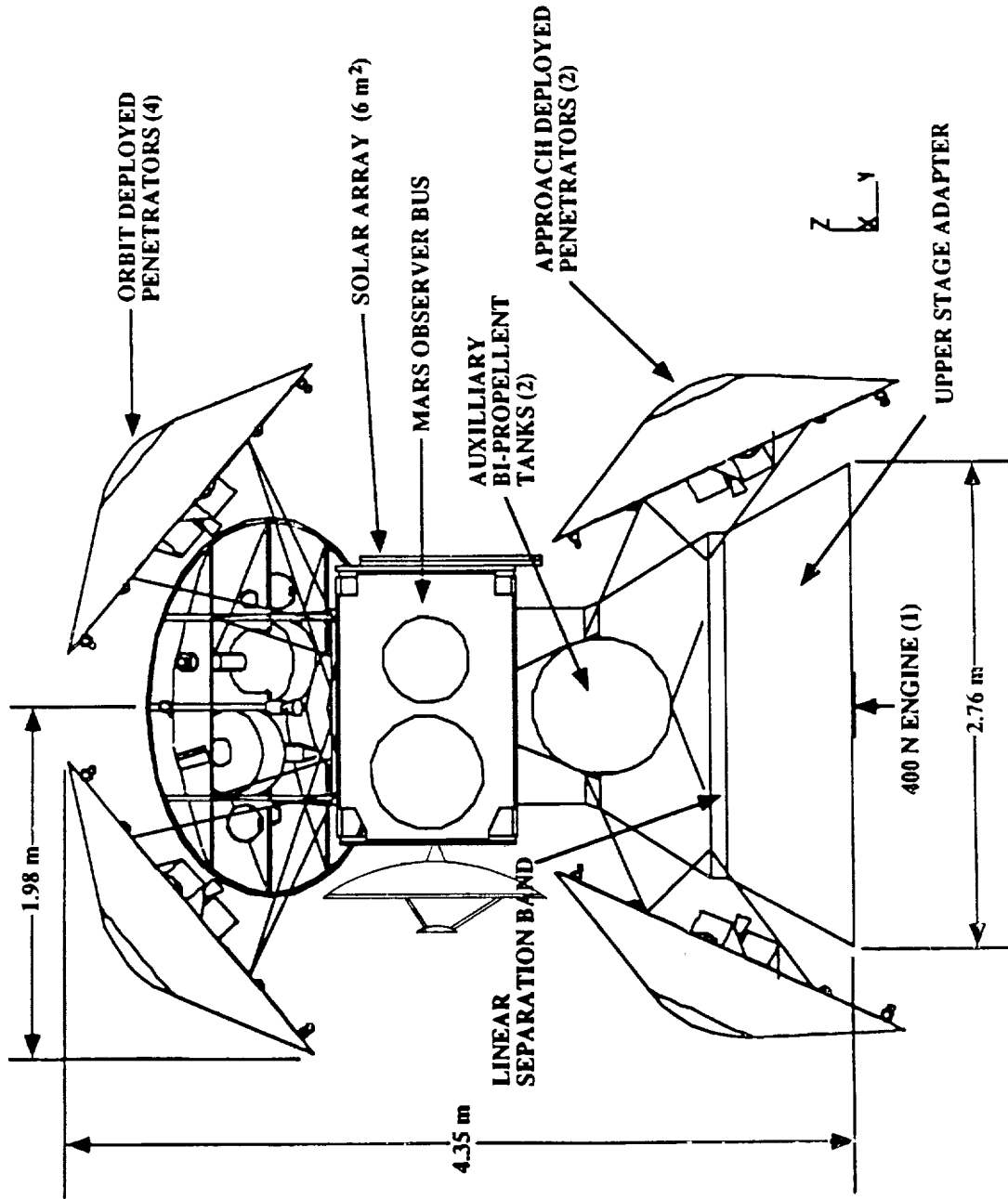


Figure 1.

MSNM AEROSHELL/PENETRATOR SYSTEM

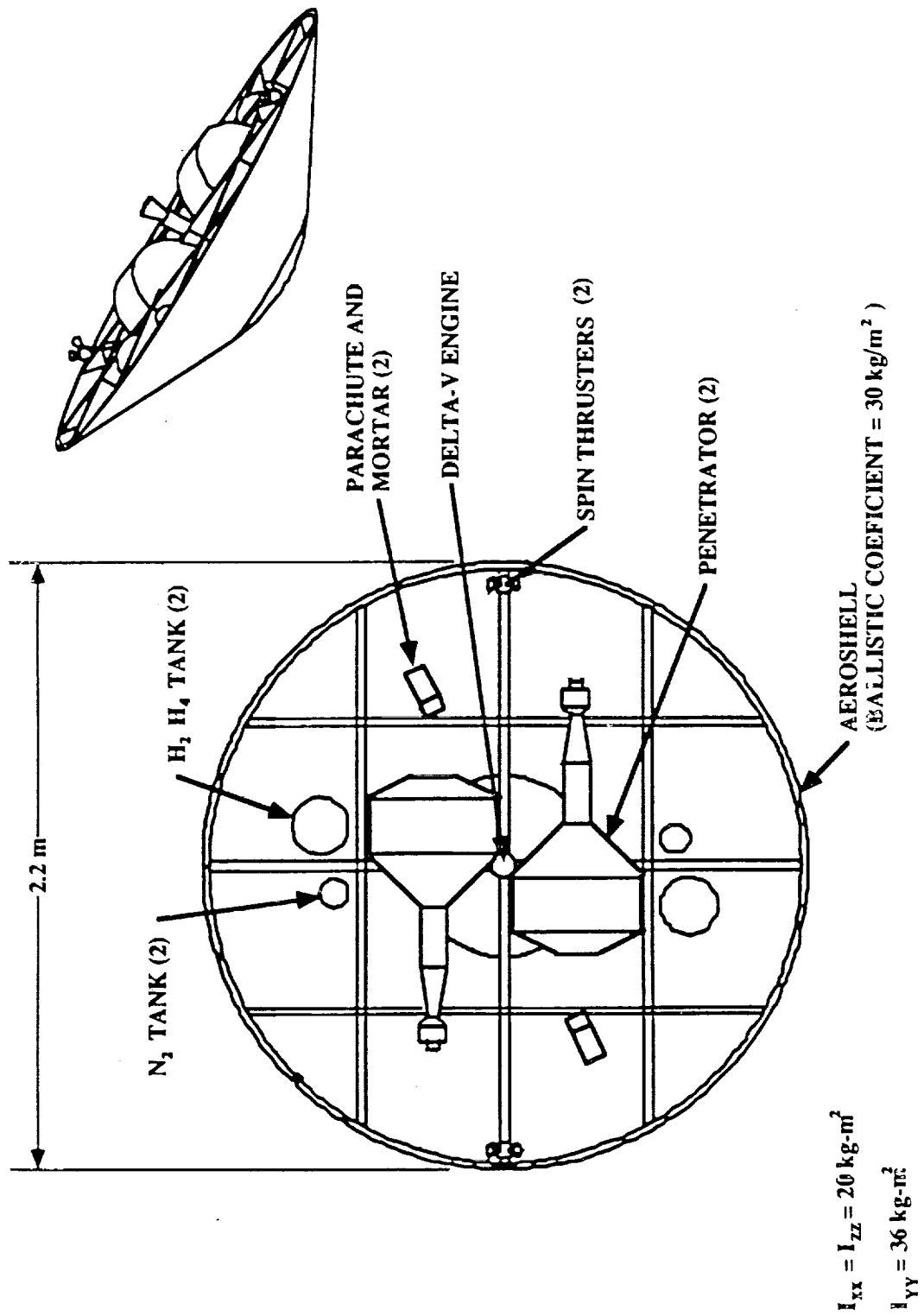


Figure 2.

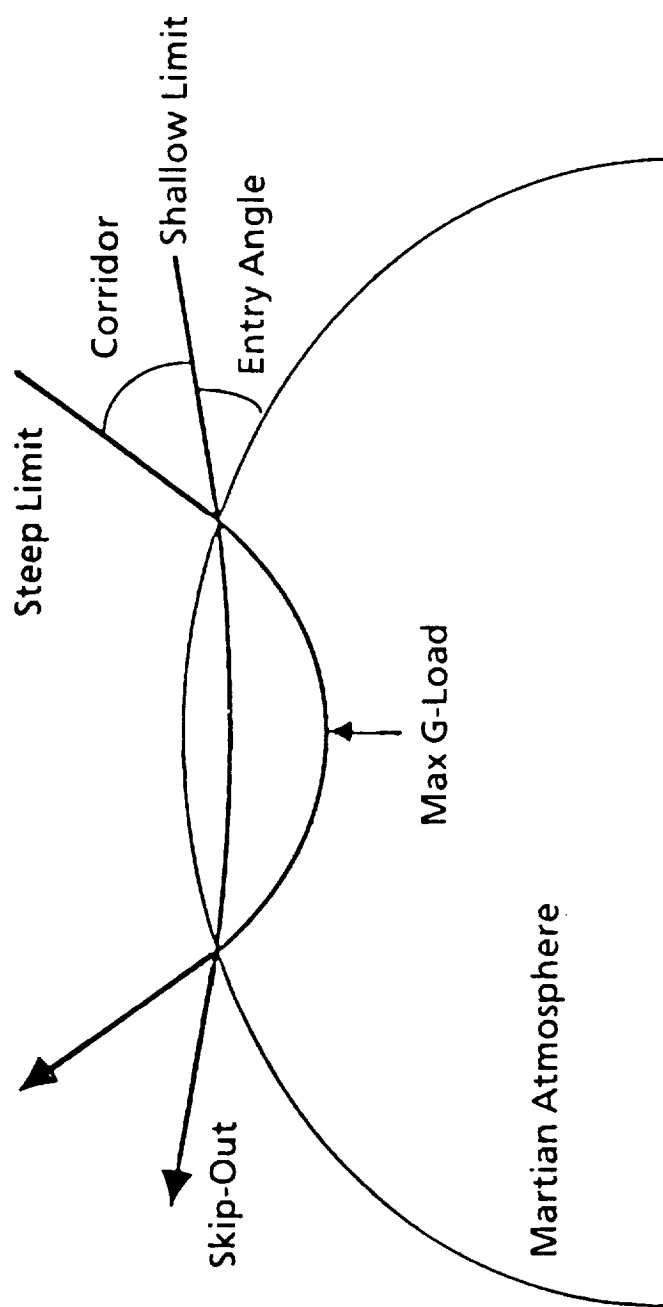


Figure 3.

ENTRY CORRIDOR LIMITS

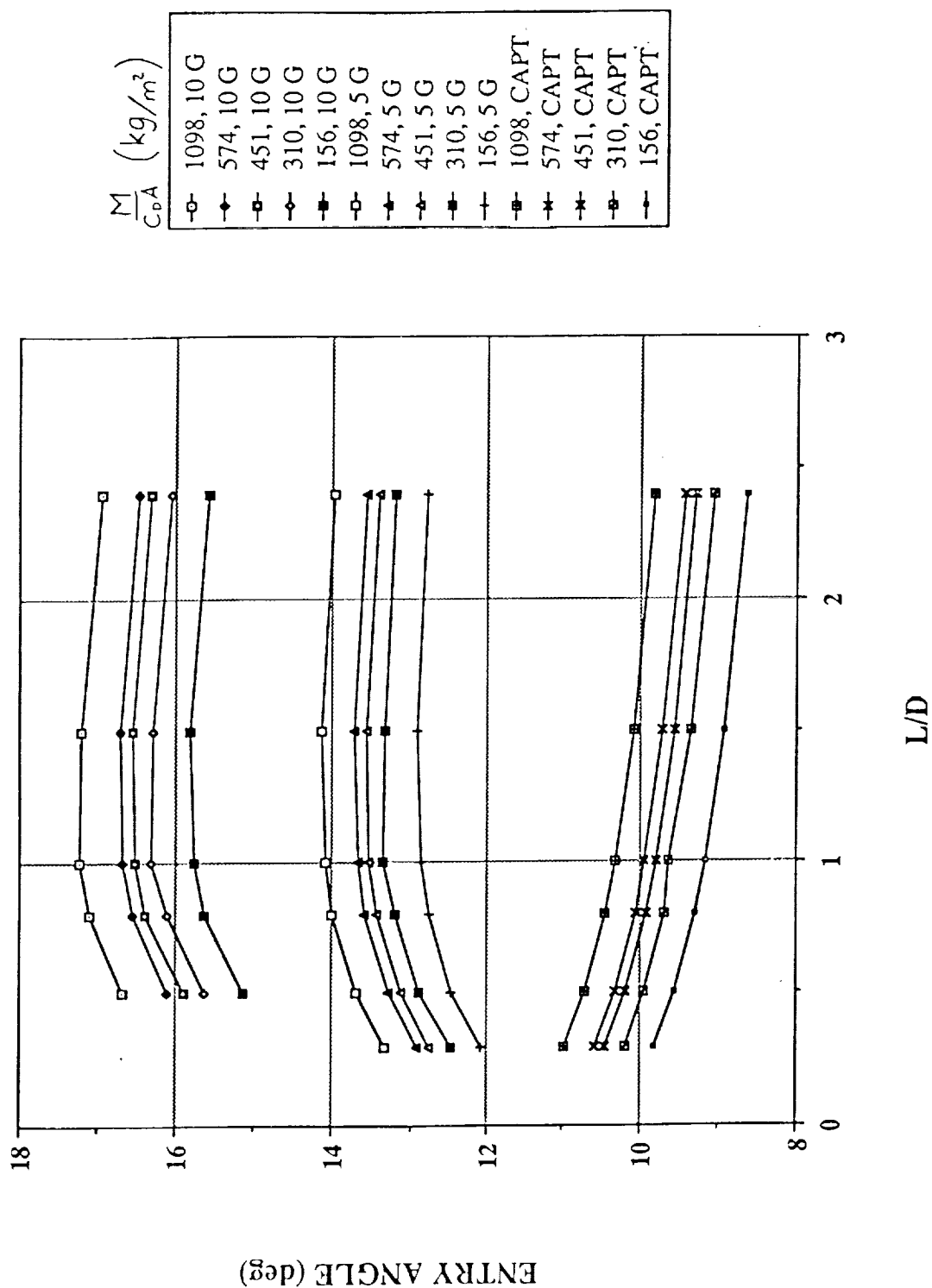


Figure 4.

EFFECTS ON CORRIDOR WIDTH

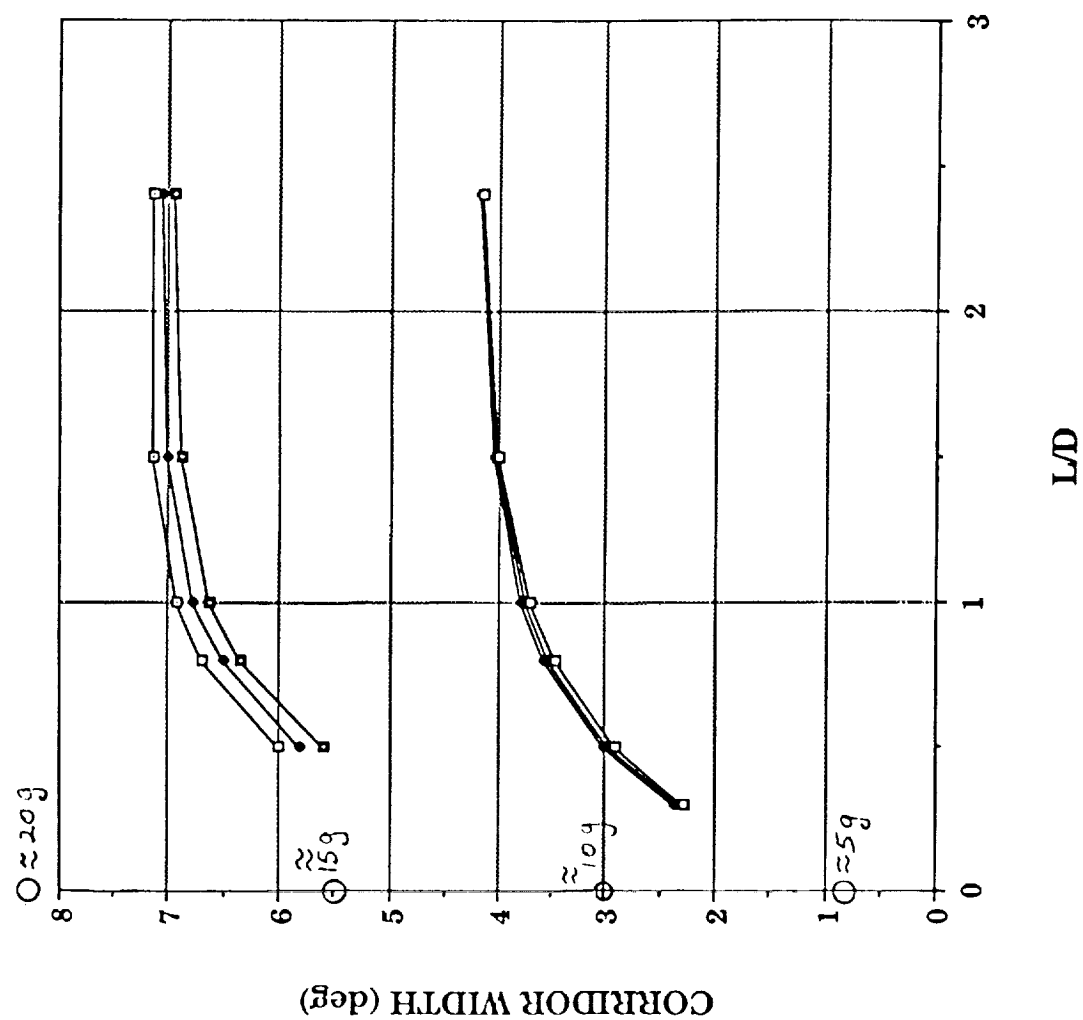


Figure 5.

G LOAD VS ENTRY ANGLE M/CDA=30 KG/M2 6 km/sec ENTRY VELOCITY

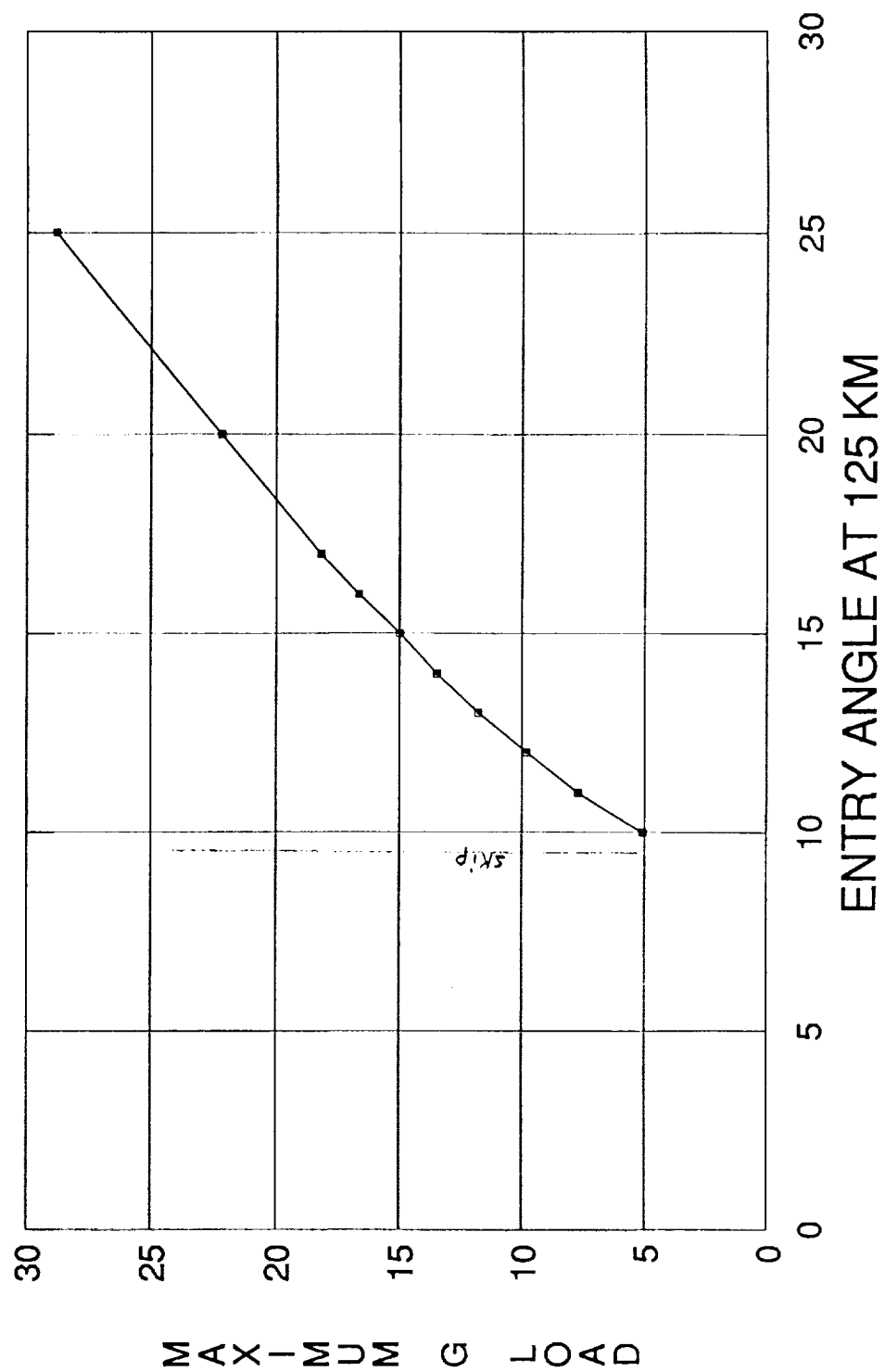


Figure 6.

RANGE VS ENTRY ANGLE M/CDA=30 KG/M2
6 km/sec ENTRY VELOCITY

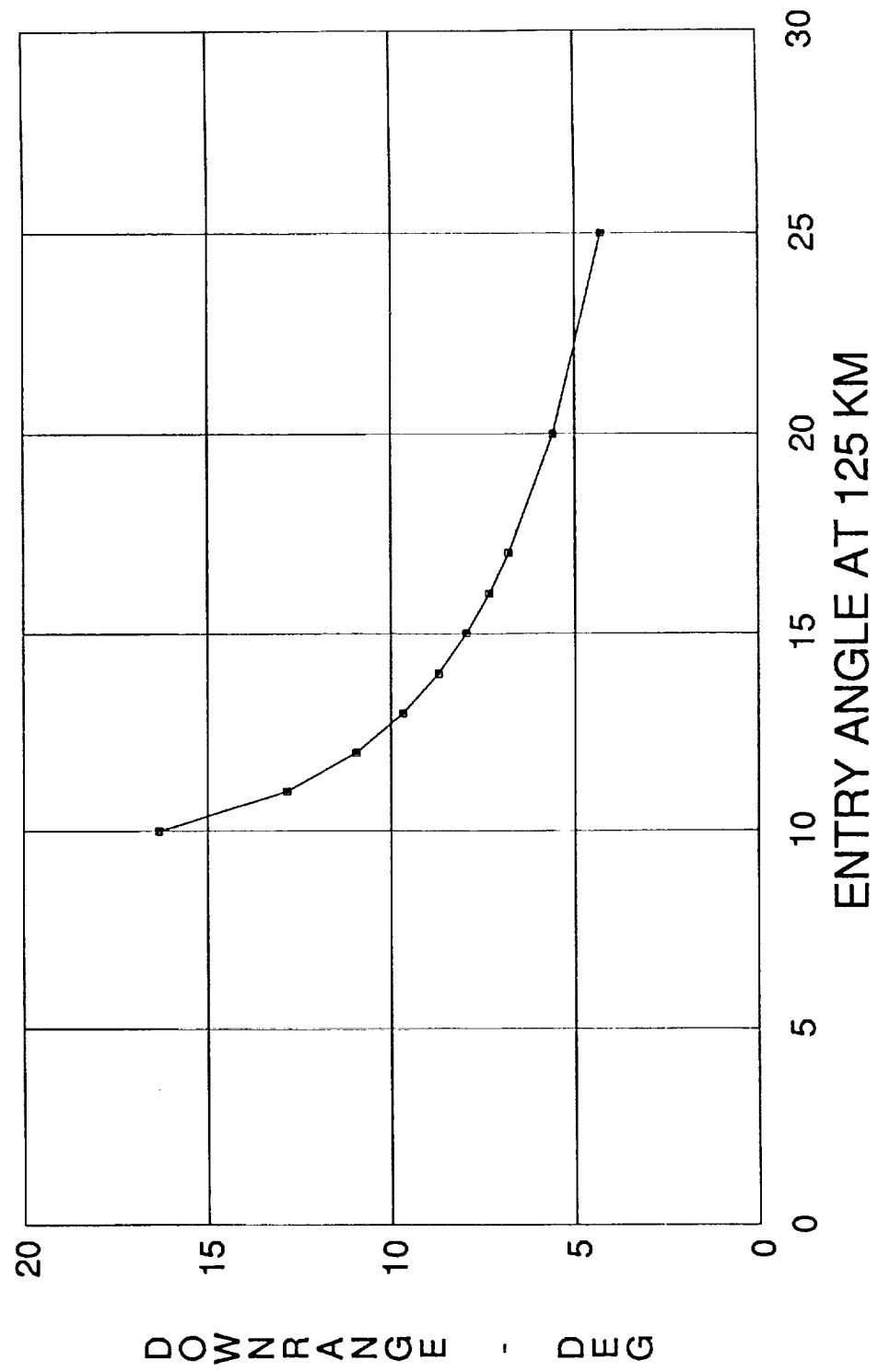


Figure 7.

RANGE VS M/CDA FOR STEEP ENTRY ANGLES 6.0 km/sec ENTRY VELOCITY

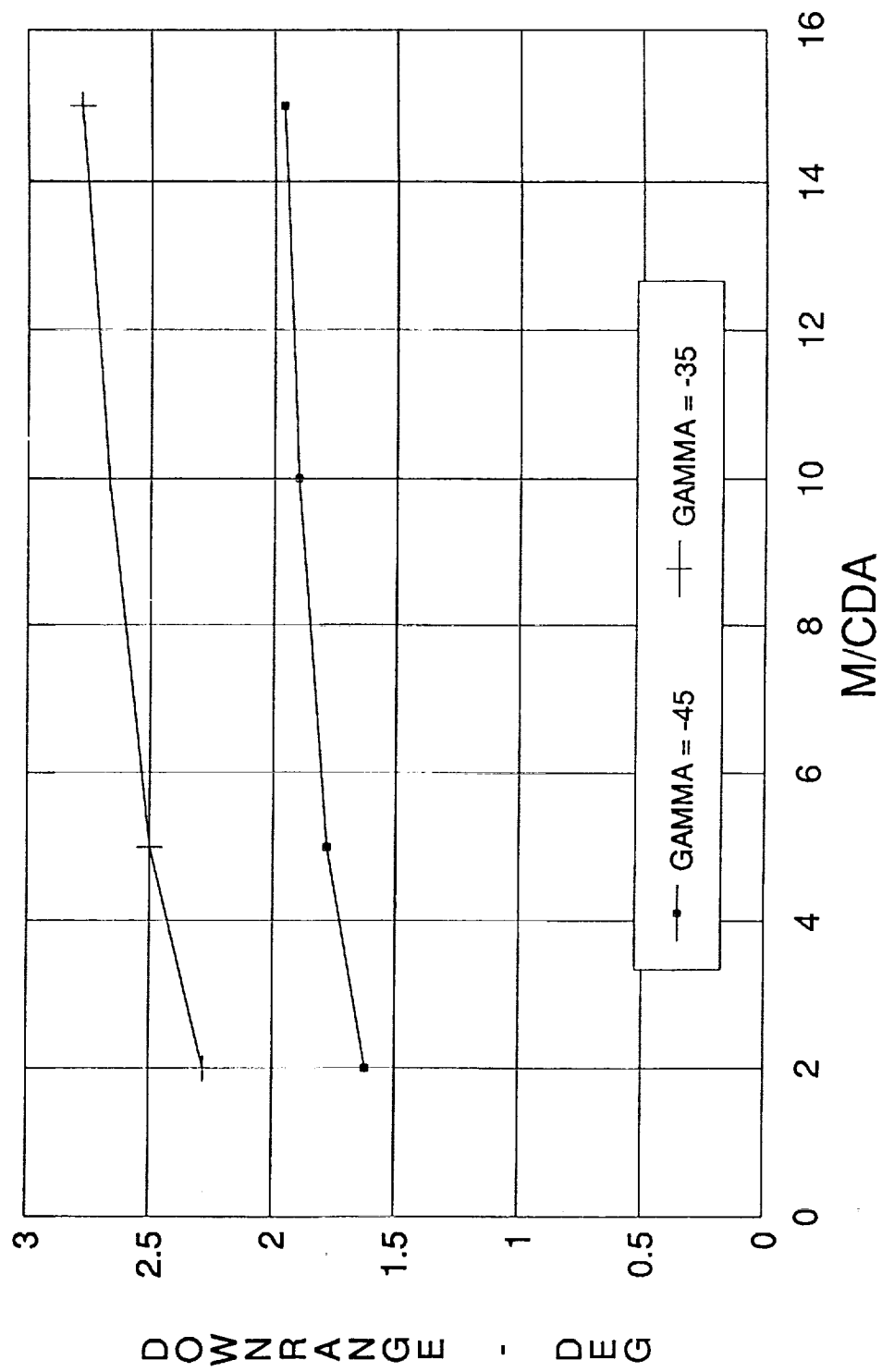


Figure 8.

G LOAD VS BALLISTIC NO. AND ENTRY ANGLE 6 km/sec ENTRY VELOCITY

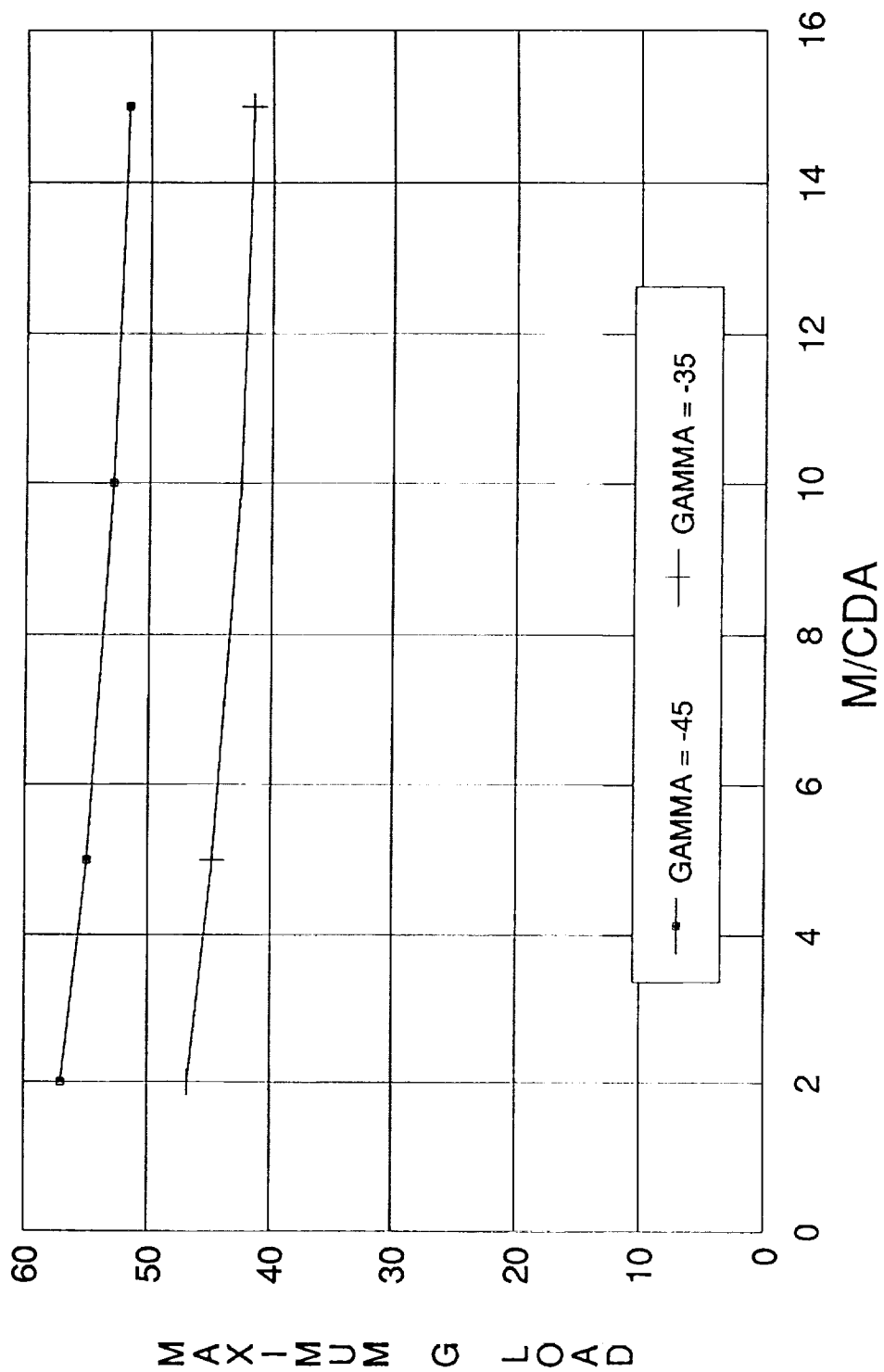


Figure 9.

G LOAD VS ENTRY ANGLE M/CDA=30 KG/M2 3.6 km/sec ENTRY VELOCITY

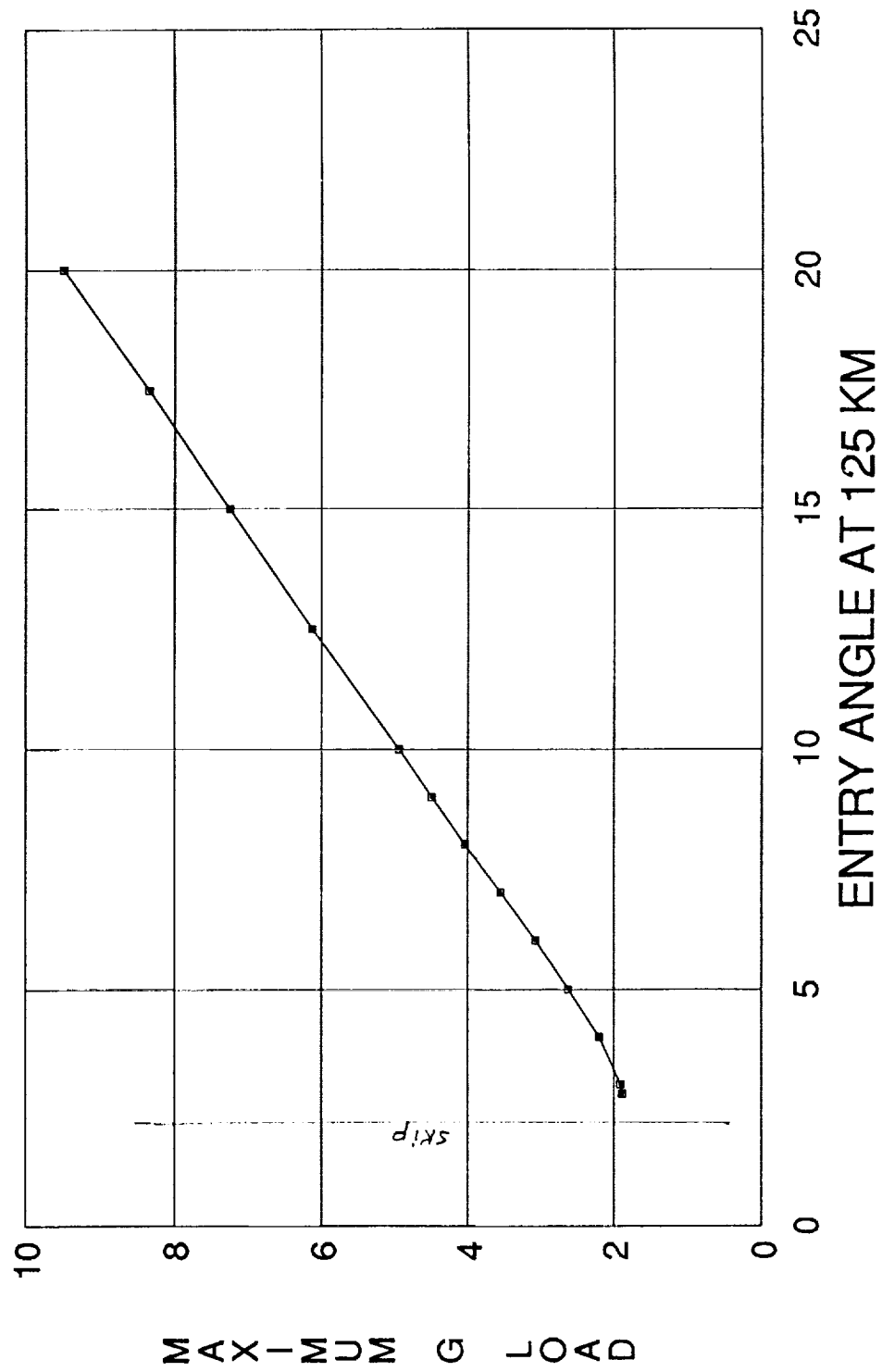


Figure 10.

RANGE VS ENTRY ANGLE M/CDA=30 KG/M2 3.6 km/sec ENTRY VELOCITY

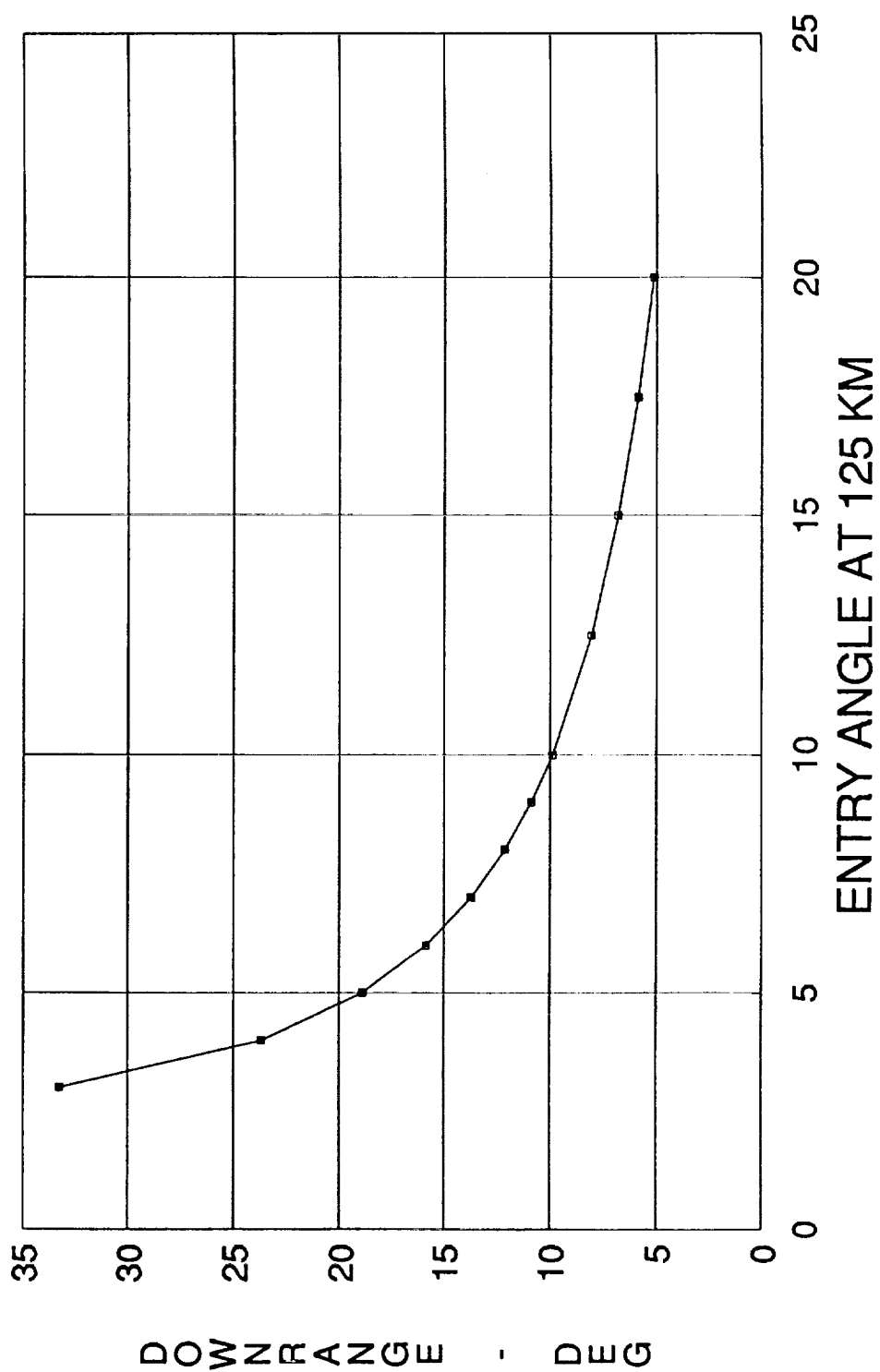


Figure 11.

VEL AT CHUTE DEPLOY VERSUS ENTRY ANGLE 3.6 km/sec ENTRY VELOCITY, M/CDA=30 KG/M

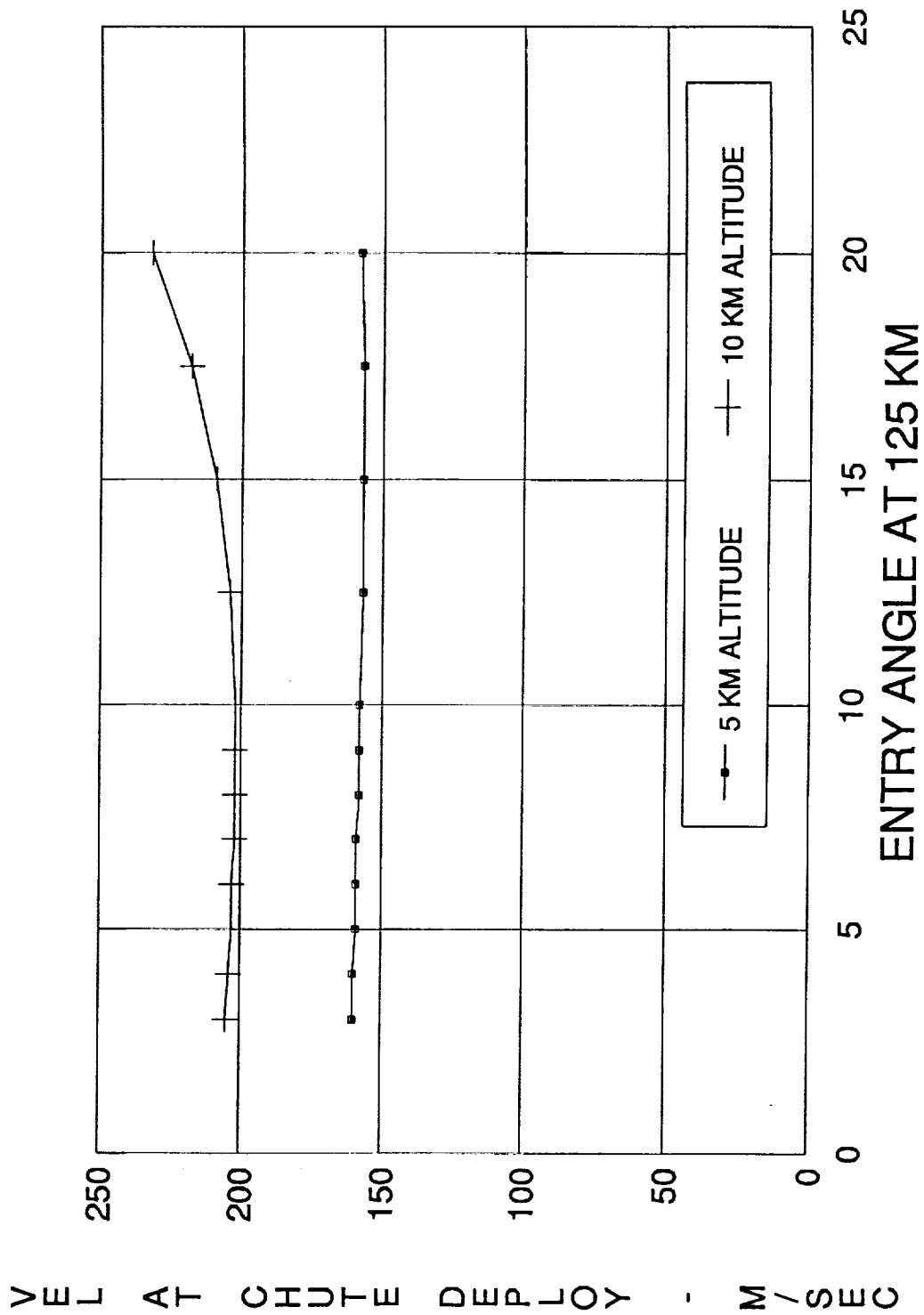


Figure 12.

VELOCITY AT 10 KM VS M/CDA AND ATMOS 3.6 KM/SEC ENTRY VELOCITY

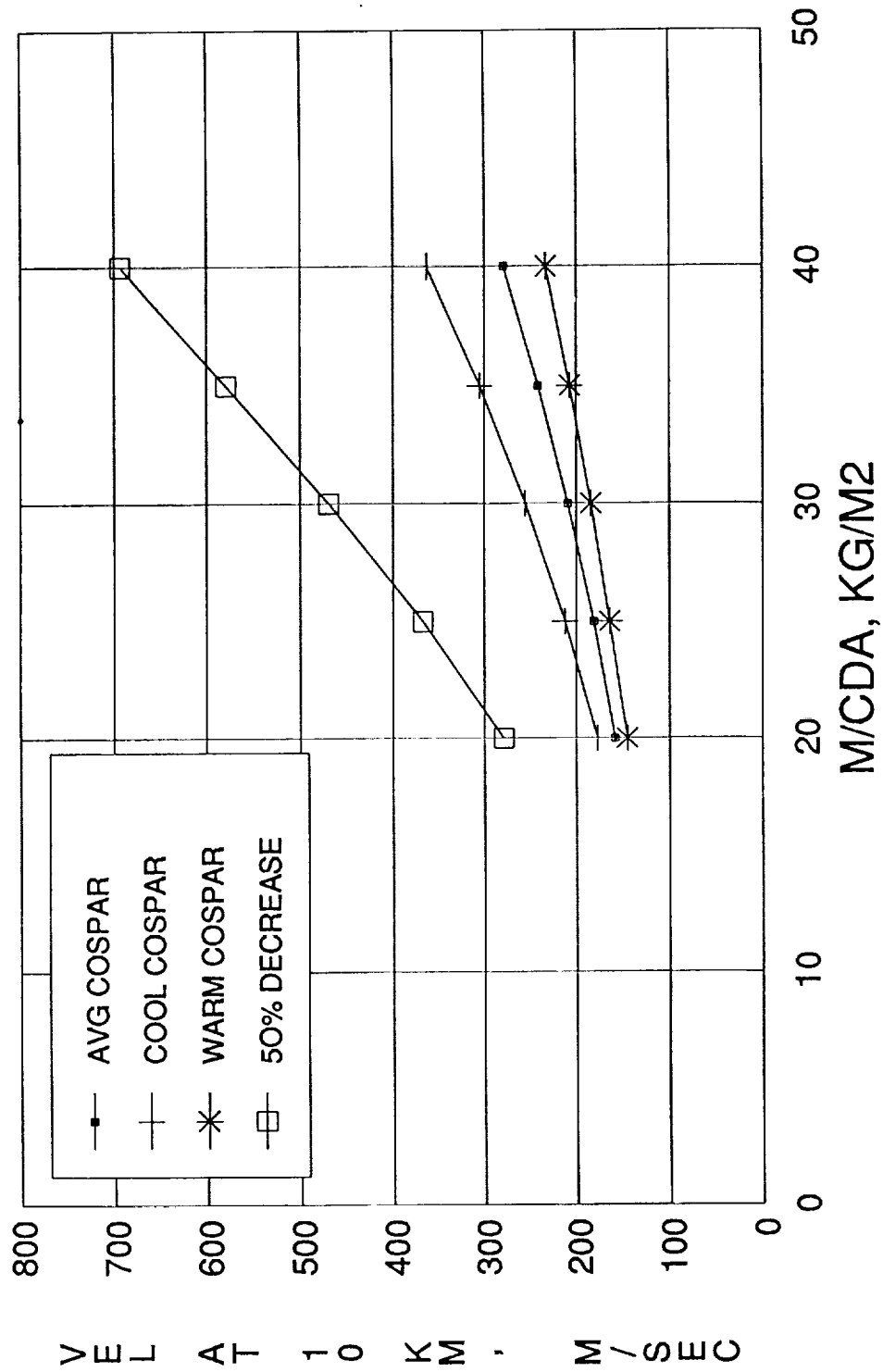


Figure 13.

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Session B, Submittal No. 9

Byron L. Swenson
Science Applications International Corporation

MARS GLOBAL NETWORK MISSION

ENTRY AND TERMINAL DESCENT SYSTEM DESIGN CONSIDERATIONS & ISSUES

**PRESENTATION TO THE
MARS GLOBAL NETWORK MISSION WORKSHOP
FEBRUARY 6-7, 1990**

BY

BYRON L. SWENSON



LANDER ENTRY SYSTEM DESIGN

REQUIREMENTS

- TO PROVIDE STABLE & CONTROLLED ATMOSPHERIC DECELERATION FROM HYPERBOLIC OR ORBITAL ENTRIES
- TO PROVIDE PROTECTION FROM AEROTHERMODYNAMIC HEATING
- TO DECELERATE LANDER TO TERMINAL DESCENT SYSTEM DEPLOYMENT CONDITIONS

DESIGN CONSIDERATIONS & ISSUES

- ENTRY CONDITIONS - SPEED, FLIGHT-PATH ANGLE, ANGLE OF ATTACK
- MARTIAN ATMOSPHERE - LOW DENSITY, VARIATIONS, SITE ELEVATIONS
- FREE-SPACE & ATMOSPHERIC DYNAMICS
- REQUIRED DELIVERY ACCURACIES
- BALLISTIC OR LIFTING TRAJECTORIES
- AERODYNAMIC & AEROTHERMODYNAMIC DATA BASE
- LOW BALLISTIC COEFFICIENT - TERMINAL SYSTEM DEPLOYMENT
- SPACECRAFT INTERFACE - TARGETING, SEPARATION, ENVELOPE
- AEROSHELL JETTISON



ENTRY SYSTEM OPTIONS

CONFIGURATIONS

- BALLISTIC
 - BLUNTED CONES - VIKING; PY, GALILEO; DISCOVERER
- LIFTING
 - SYMMETRICAL (C.G. OFFSET) - BICONICS
 - NON-SYMMETRIC - RAKED CONES (AFE); HALF-CONES (M-1)

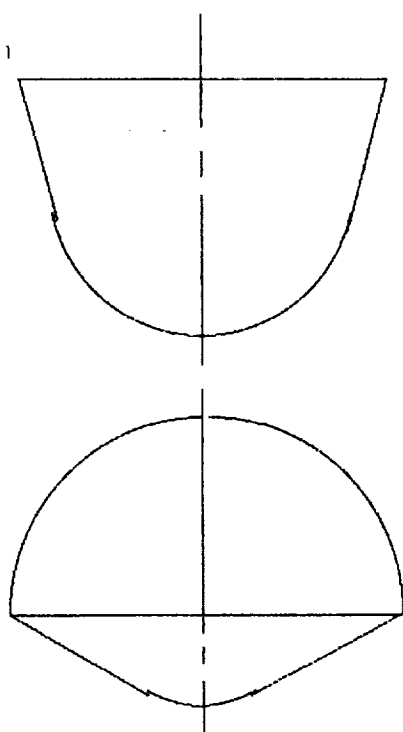
AEROSHELL DESIGN

- RIGID
 - SKIN/STRINGER; ABLATOR
- DEPLOYABLE
 - INFLATABLE BALLUTES - FABRIC; RERADIATOR OR ABLATOR
 - COLLAPSABLE - FABRIC/RIB & STRUT; RERADIATOR OR ABLATOR
 - FOLDABLE - RIGID HINGED PANELS; RERADIATOR OR ABLATOR



CONFIGURATION OPTIONS

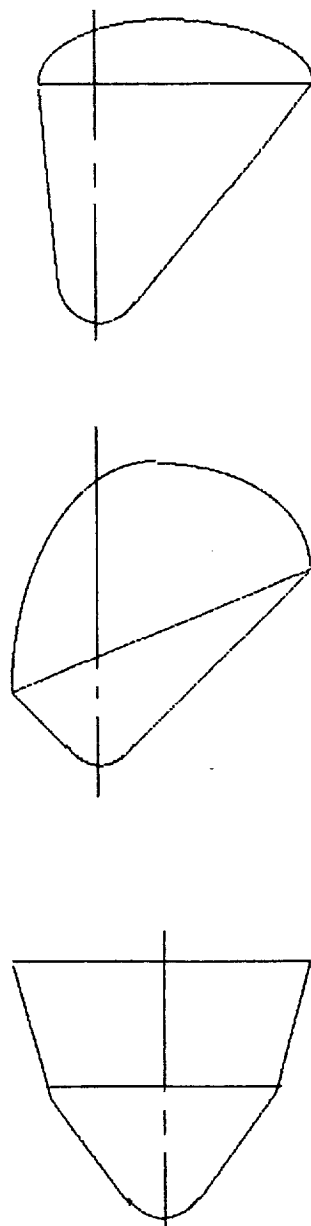
BALLISTIC



VIKING

DISCOVERER

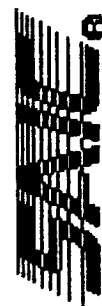
LIFTING



BICONIC

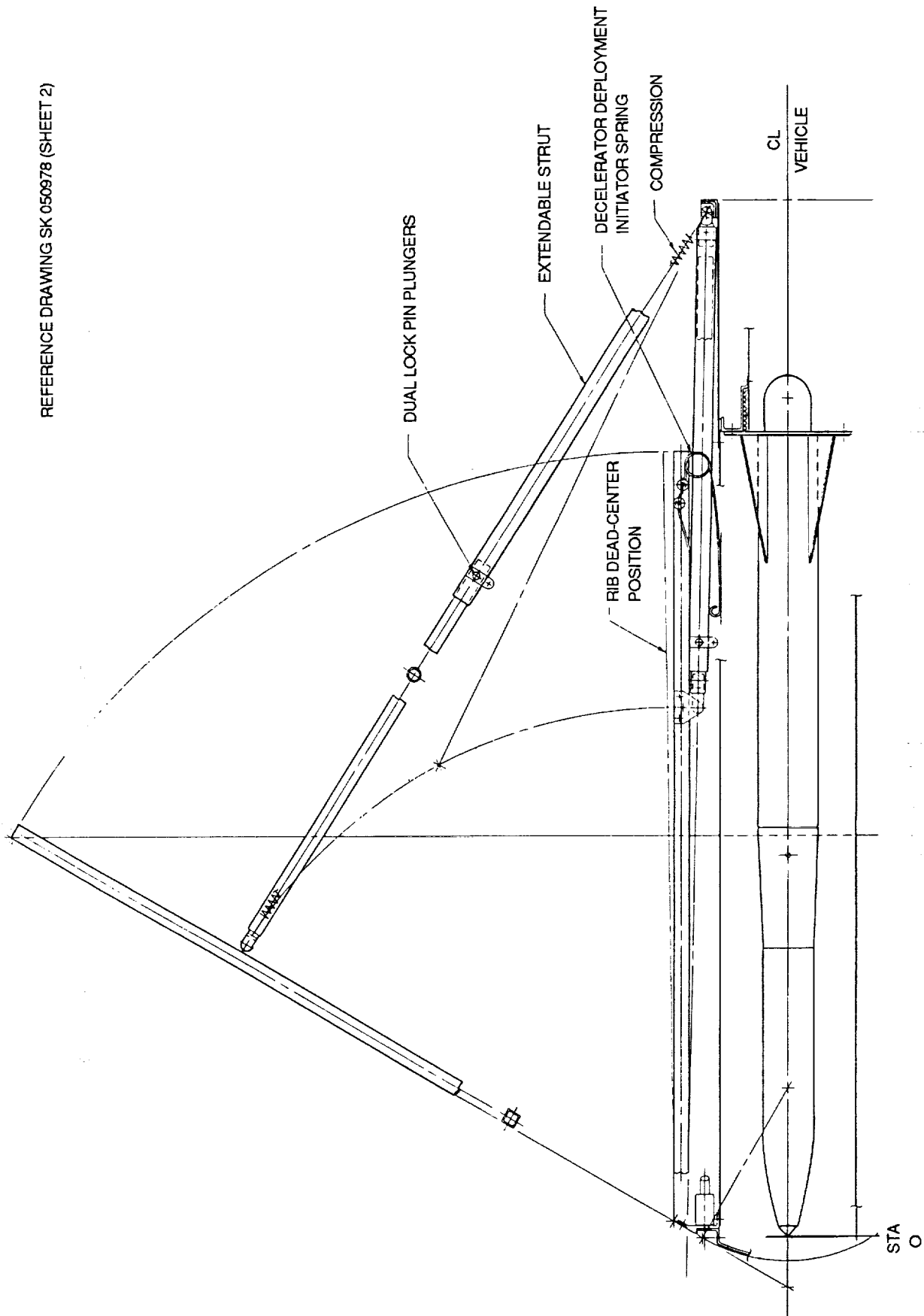
AFE

M-1



MARS PENETRATOR - EXTENDIBLE STRUT DECELERATOR

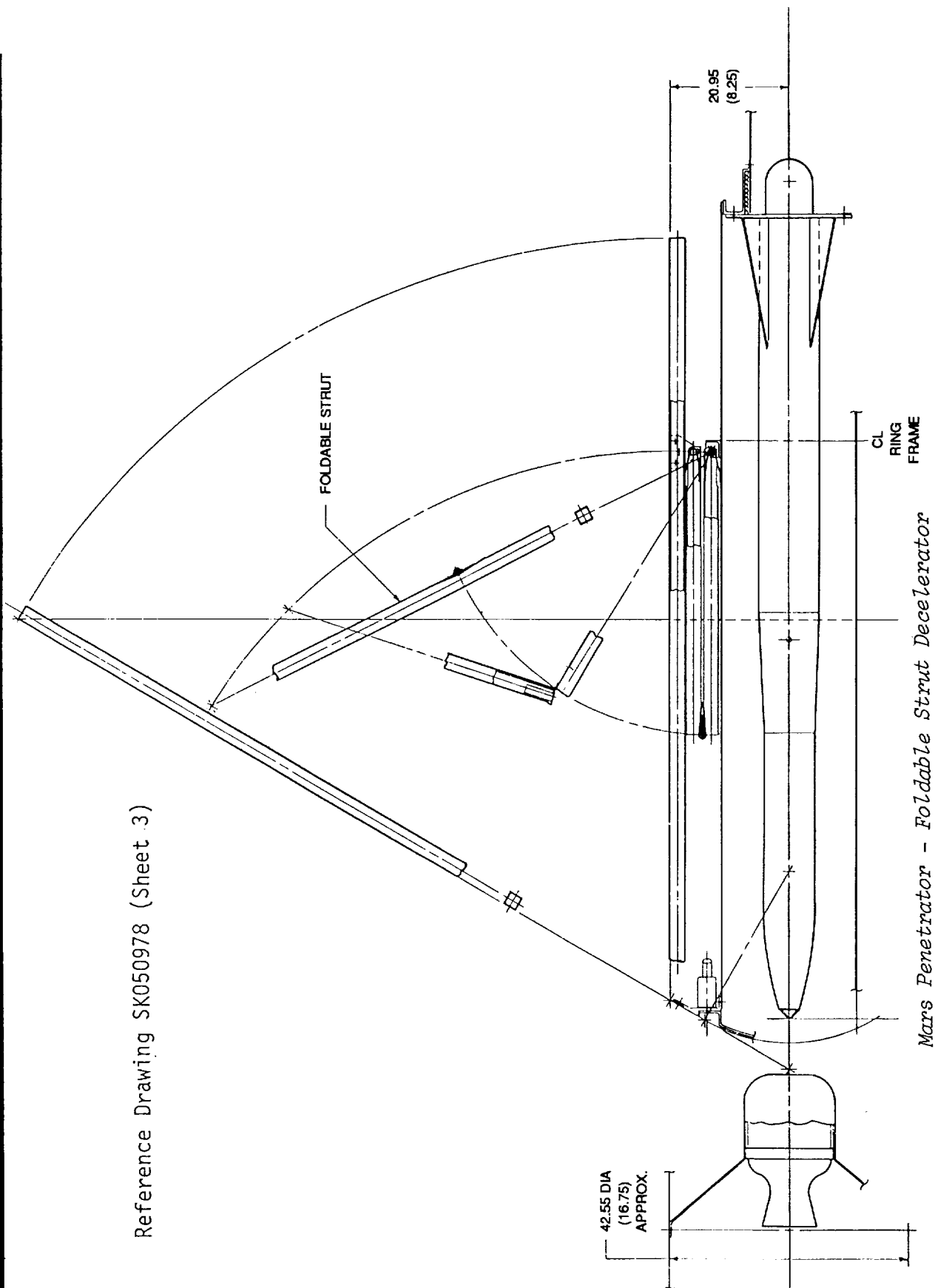
REFERENCE DRAWING SK 050978 (SHEET 2)



Mars Penetrator - Extendible Strut Decelerator

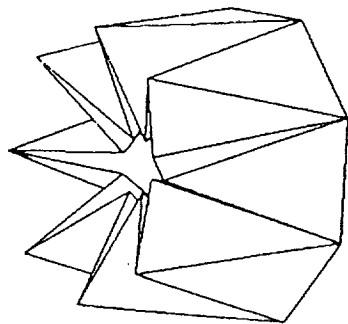
MARS PENETRATOR - FOLDABLE STRUT DECELERATOR

Reference Drawing SK050978 (Sheet 3)

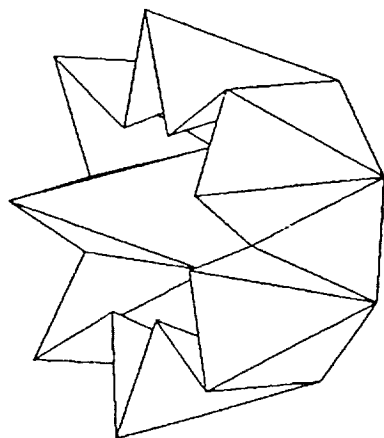


HINGED PANEL CONCEPT FOR DEPLOYABLE DECELERATOR

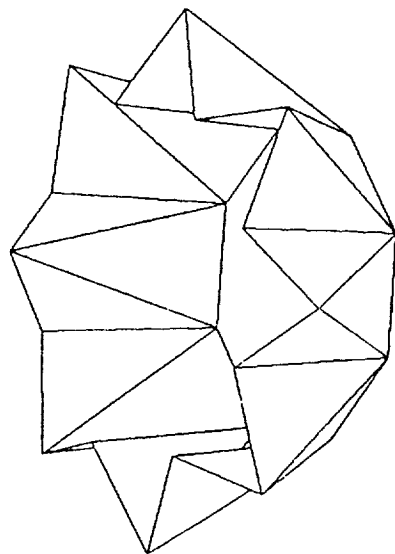
STOWED



A



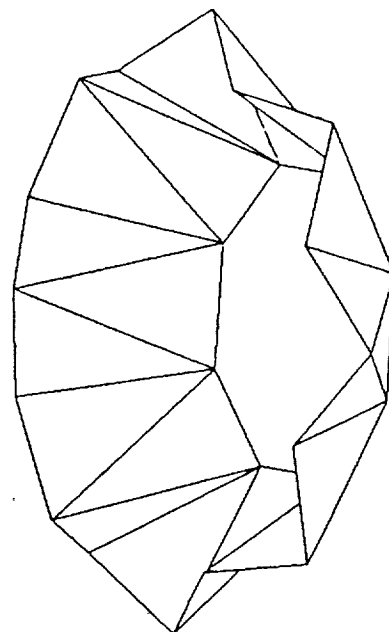
B



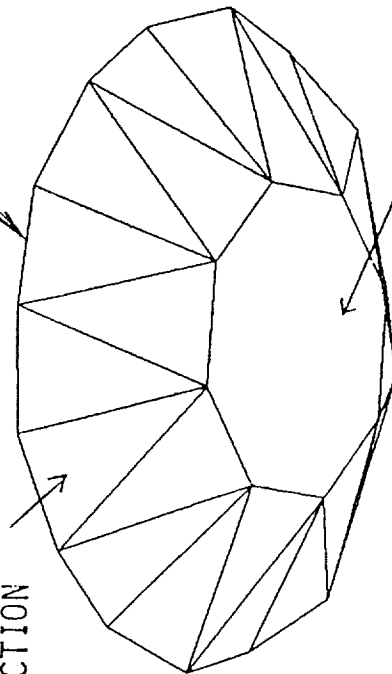
C

FULLY
DEPLOYED

TRIANGULAR SKIRT
SECTION



D



E

NOSECAP

AEROSHELL JETTISON OPTIONS

- **PULL OUT OF THE REAR**
 - **REQUIRES LARGE PARACHUTE SYSTEM**
- **FALL-THRU THE NOSE**
 - **CONSTRAINS LANDER SHAPE**
 - **NOSE CAP JETTISON**
- **SEPARATE IN PIECES**
 - **HAZARDOUS**
 - **REQUIRES EXTENSIVE PYROTECHNICS AND VERY ACCURATE TIMING**
- **RETAIN TO LANDING**
 - **LANDING SYSTEM DESIGN IMPACT**



TERMINAL DESCENT SYSTEM DESIGN

REQUIREMENTS

- TO PROVIDE A STABLE & CONTROLLED PLATFORM FOR DESCENT IMAGERY
- TO PROVIDE STABLE & CONTROLLED SURFACE APPROACH CONDITIONS FOR LANDING

DESIGN CONSIDERATIONS & ISSUES

- DEPLOYMENT CONDITIONS
- TECHNOLOGY BASE
- DESCENT TIME, RATE, & STABILITY FOR IMAGERY
- WINDS ALOFT & SHEARS
- ELEVATION DIFFERENCES OF SITES
- REQUIRED LANDING CONDITIONS
 - VERTICAL & HORIZONTAL VELOCITY
 - STABILITY & DYNAMICS
- SURFACE WINDS & SHEARS



Session B, Submittal No. 10

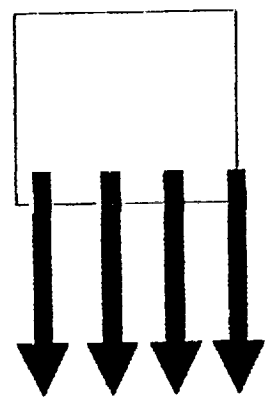
Arden Albee
California Institute of Technology

COMPARISON OF ALTERNATIVE LANDER STRATEGIES

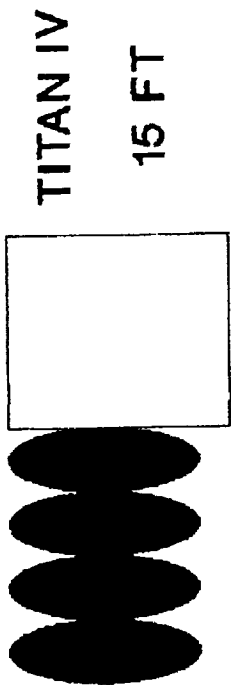
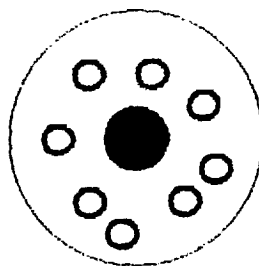
ARDEN ALBEE

CALTECH

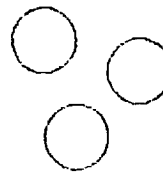
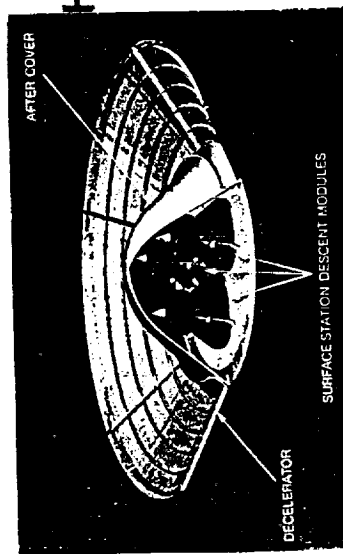
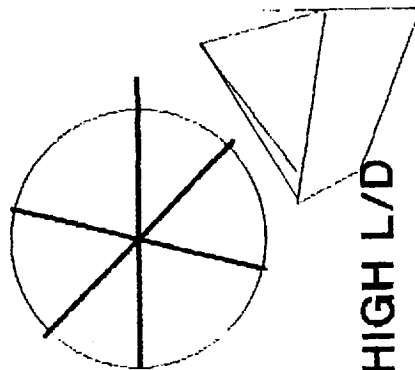
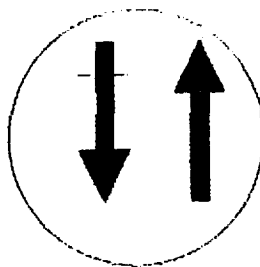
"MANY, GLOBALLY-DISPERSED" LANDERS DRIVE DESIGN PACKING AND DEPLOYMENT VS STRUCTURE/PROP MASS



INDIVIDUAL ENTRY



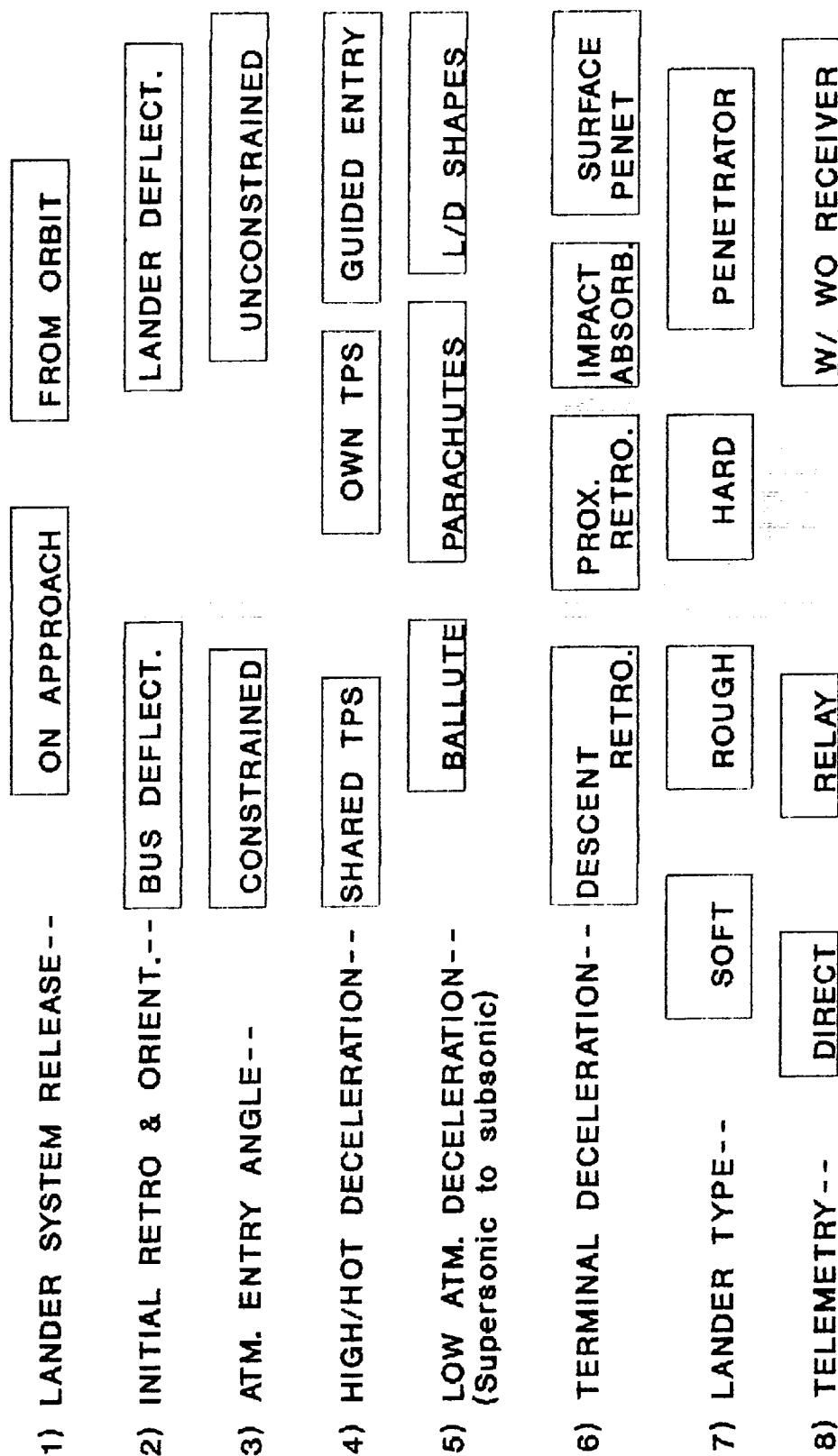
SHARED ENTRY INDIV. ENTRY



SMALL AEROSHELLS

MARS ENTRY & LANDER SYSTEMS

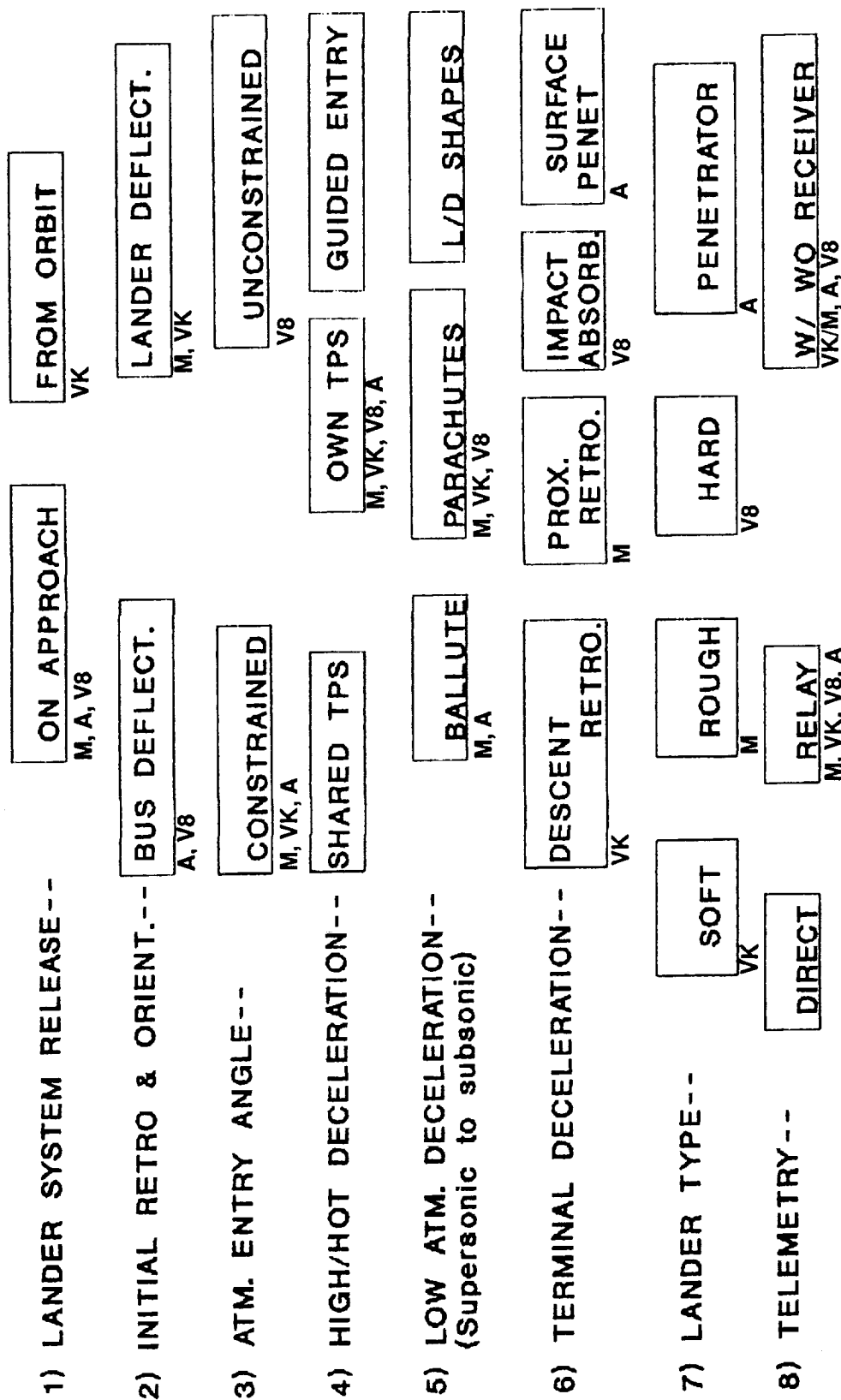
DESIGN TREE



ALA-2-90

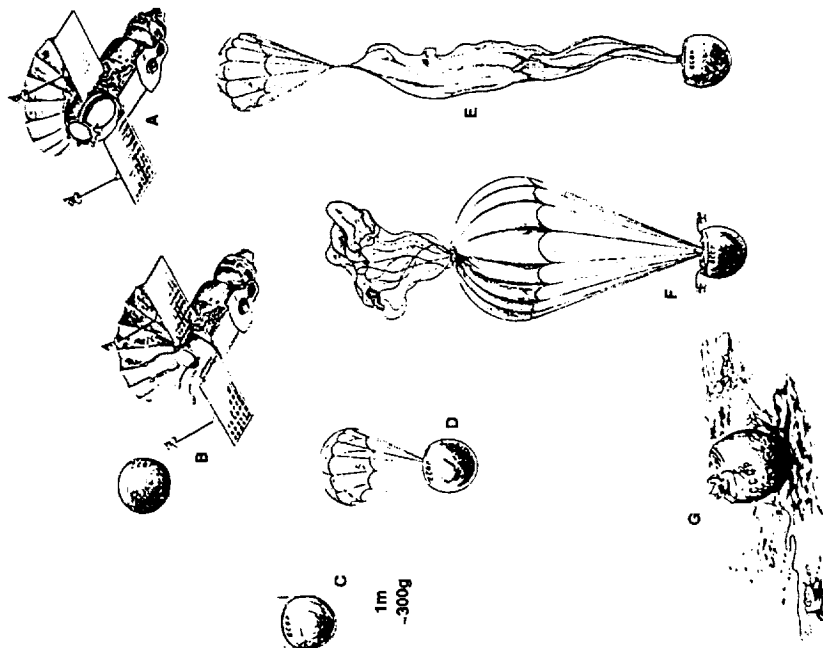
MARS ENTRY & LANDER SYSTEMS

DESIGN TREE



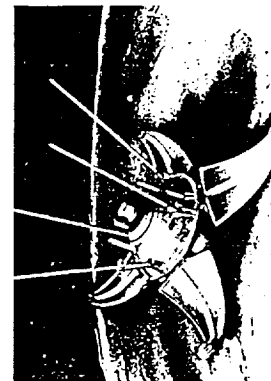
Key: M = Mars 3; VK = Viking; V8 = Venera 8; A = Ames 77

SPHERICAL ENTRY SYSTEMS

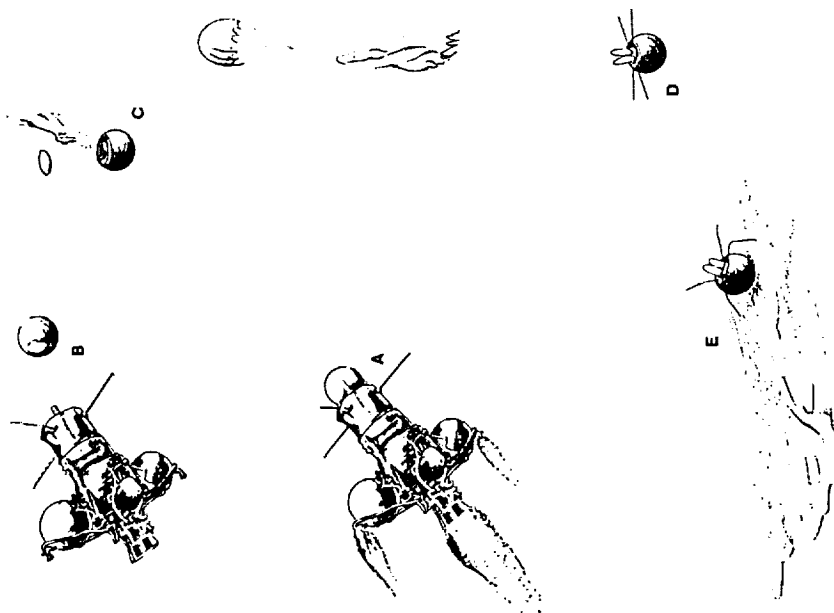


A. Venera 8 as it approaches Venus
 B. Separation of descent capsule from carrier bus
 C. Entry into Venusian atmosphere
 D. Deployment of drogue parachute
 E. Deployment of main parachute
 F. Activation of scientific instruments
 G. Landing on the Venusian surface

Descent sequence of Venera 8 (R. Escarcega)



Sketch of Luna 9 on the lunar surface

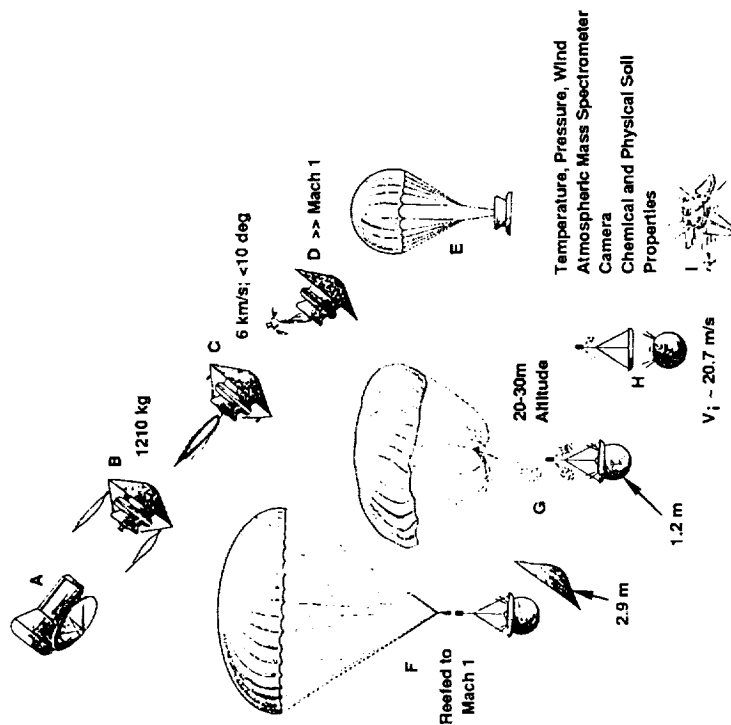
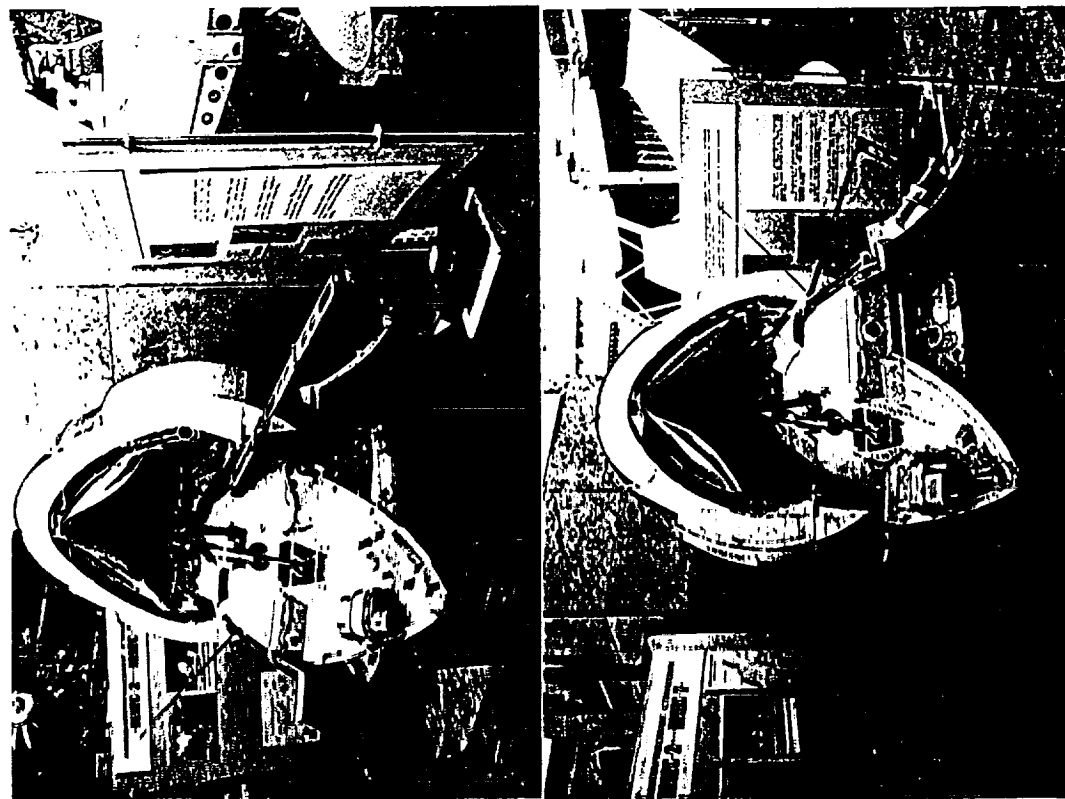


A. Blast-off from the moon
 B. Ejection of the return capsule near earth
 C. Drogue parachute deployment after reentry into earth's atmosphere; 20-g deceleration
 D. Deployment of the main parachute
 E. Touchdown of the return capsule on Soviet territory

The return of Luna 16 to earth (R. Escarcega)

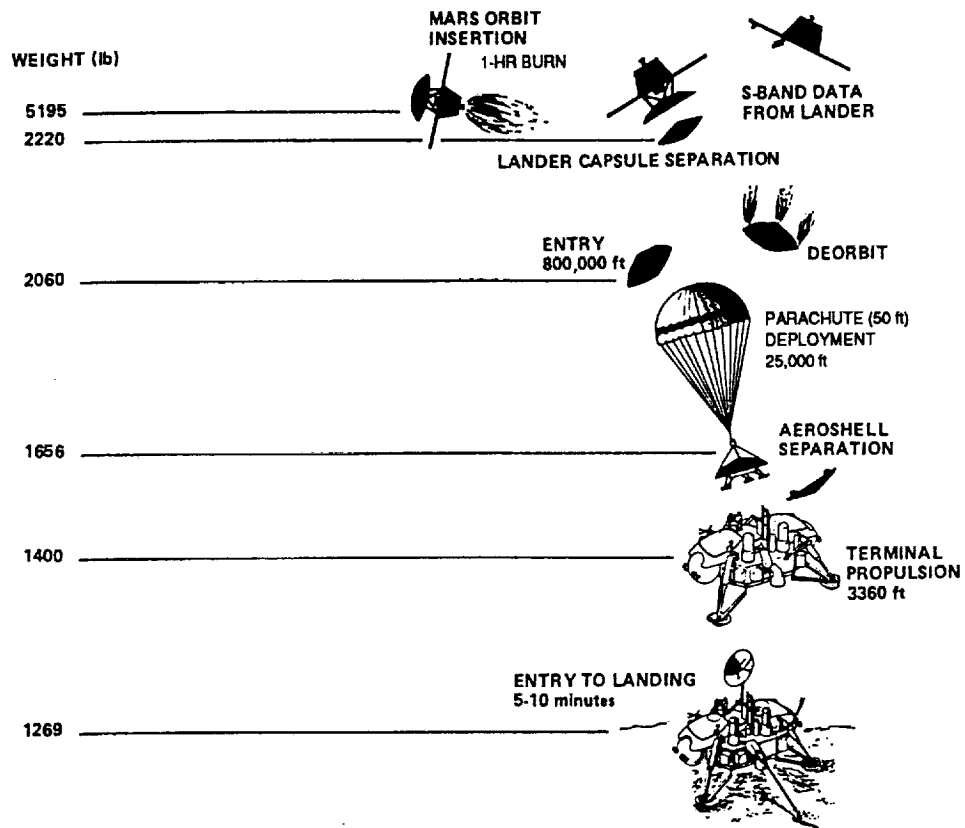
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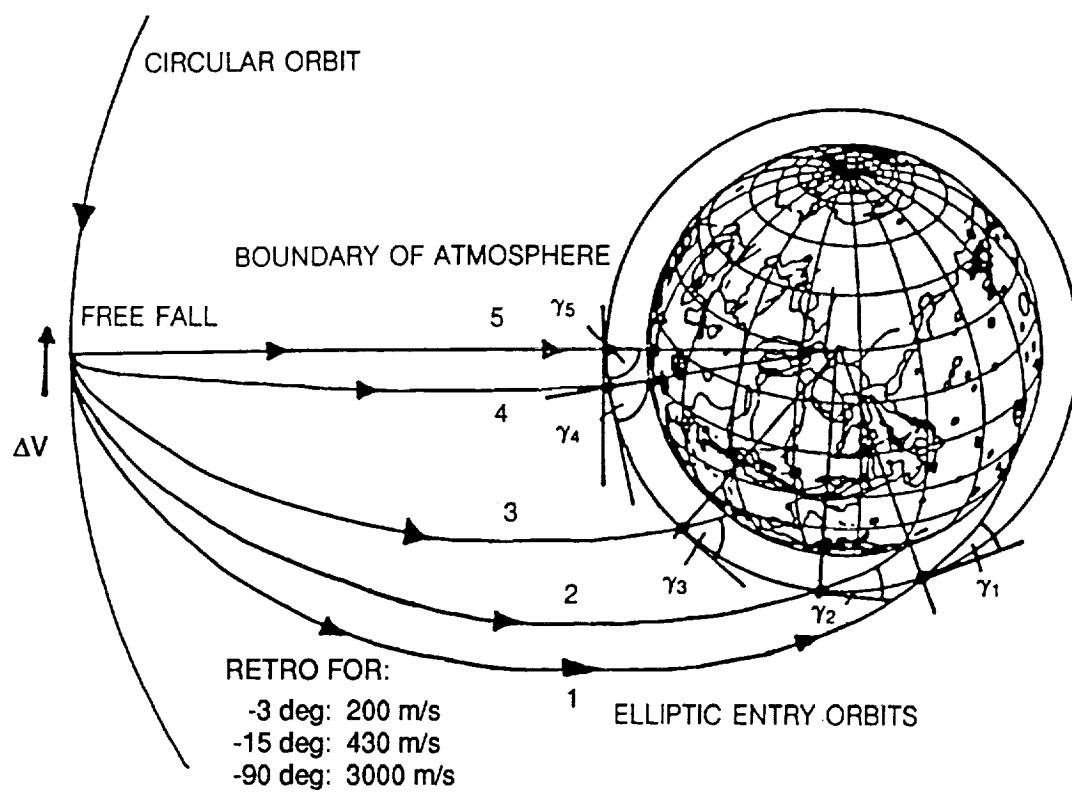
MARS 3 LANDER



- A. Mars orbiter after separation of descent apparatus
 - B. Descent apparatus pulling away from Mars orbiter -4 1/2 hr before MOI
 - C. Firing of main descent engine for entry into the Martian atmosphere
 - D. Deployment of drogue parachute
 - E. Separation of drogue parachute and upper portion of main parachute housing
 - F. Deployment of main parachute
 - G. Detachment of main parachute and ignition of final braking rocket
 - H. Separation of landing capsule
 - I. Landing capsule on the Martian surface with deployed instrumentation
- Landing sequence of second-generation Mars spacecraft
(R. Escarcega)

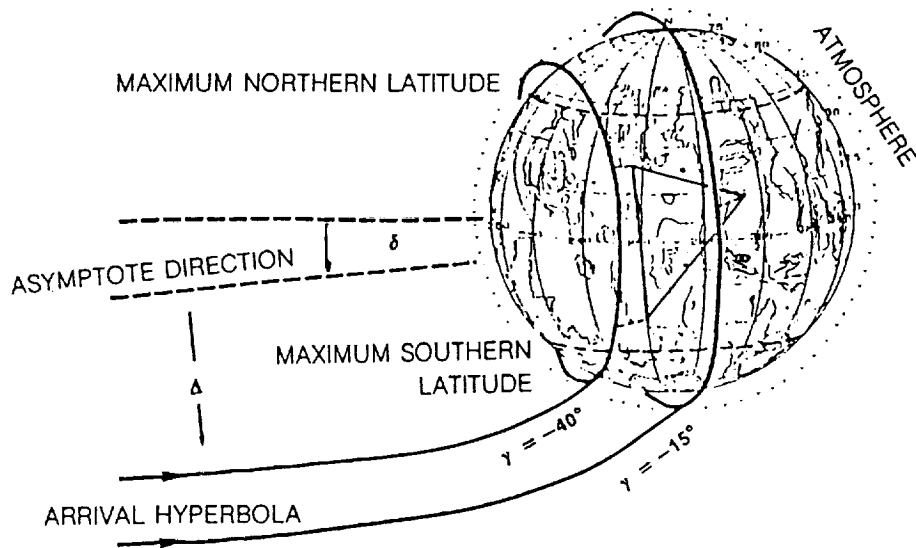
Viking Entry Through Landing Sequence



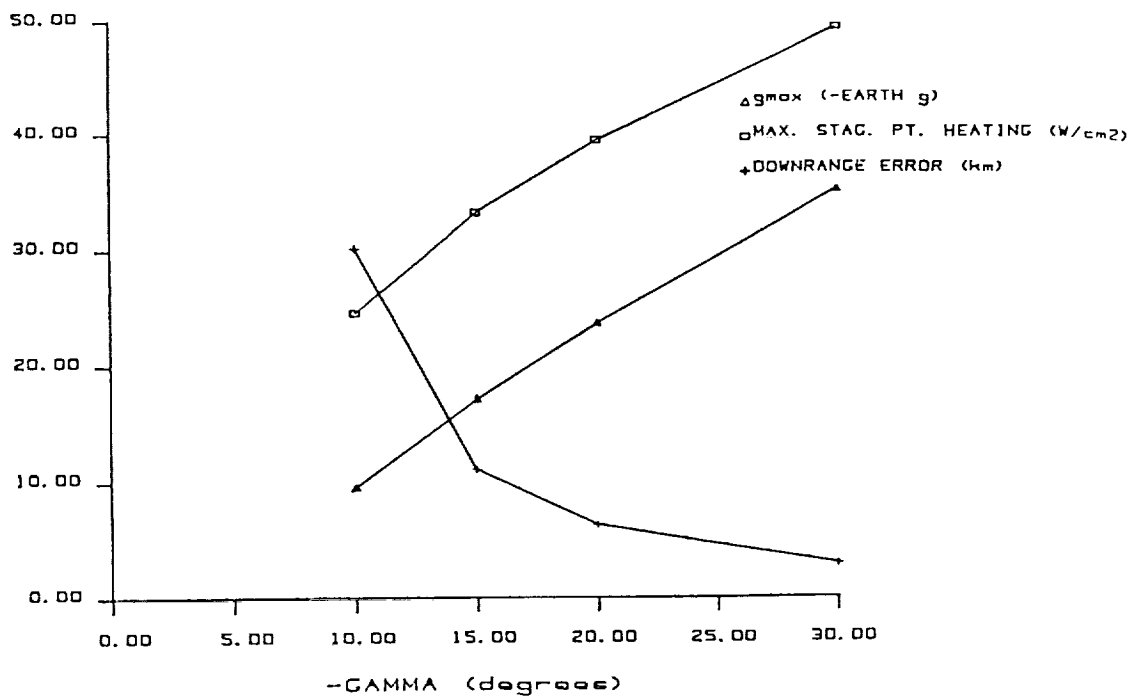


Deorbit manoeuvre from low circular orbit.

NETWORK MISSION SCENARIO



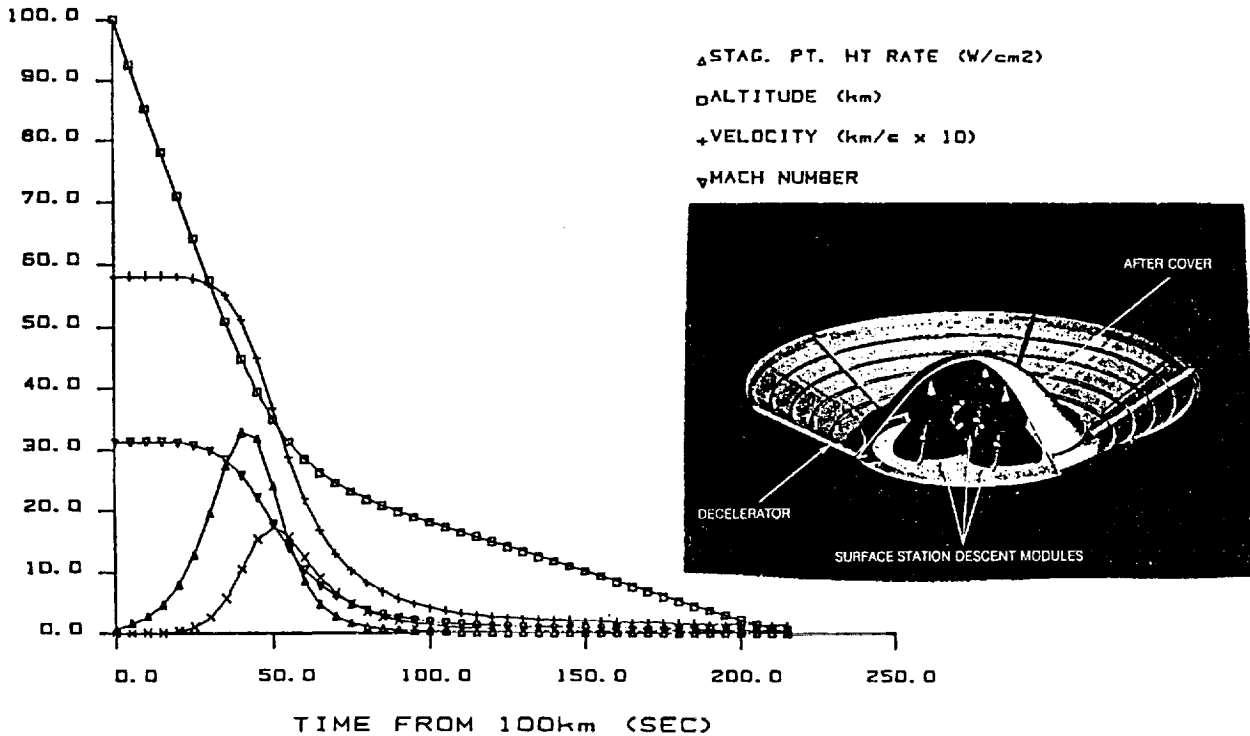
Location of entry points at Mars' atmosphere.



$R_n=1.25\text{m}$ $R_b=1.55\text{m}$ CONE ANGLE= 60° $M=300\text{kg}$

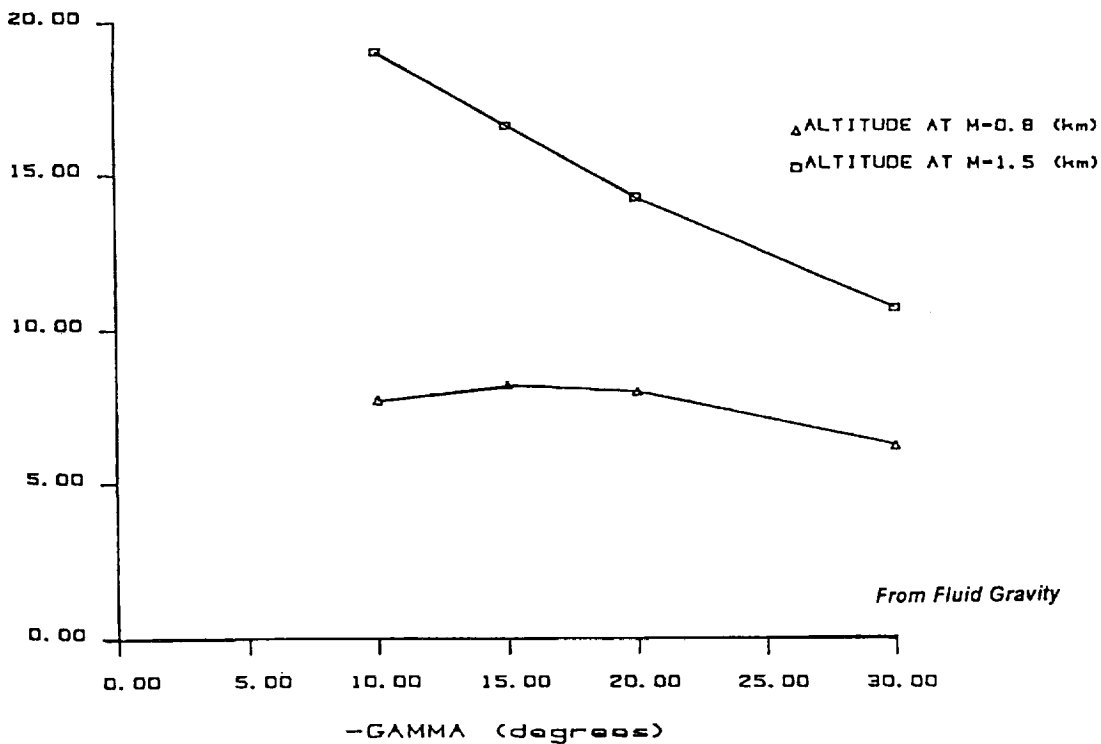
Ballistic entry from hyperbolic arrival (performance for different entry angles).

NETWORK MISSION SCENARIO



GAMMA--15 Rn=1.25m Rb=1.55m CONE ANGLE=60 M=300kg

Ballistic entry from hyperbolic arrival (trajectory parameters).



M=300kg LOW PRESSURE ATMOSPHERE

Ballistic entry from hyperbolic arrival (parachute deployment altitude).

Session B, Submittal No. 11

Lester L. Sackett
The Charles Stark Draper Laboratory

GLOBAL NETWORK MISSION WORKSHOP

Achievable Accuracies for Targeting
Landing Sites

Les Sackett (617/258-2289)
The Charles Stark Draper Laboratory
Cambridge, Massachusetts 02139

Workshop held at the Jet Propulsion Laboratory
February 6-7, 1990

Data provided by
John Higgins (617/2582433)
Ken Spratlin (617/258-2441)



Global Network Workshop

Primary Question:

What are the desired and achievable accuracies for targeting the landing sites?

Subsidiary Questions:

What are the navigation (knowledge) uncertainties at the time of aeroshell firings?

What are the landing (guidance) dispersions of the penetrators?

What contributes to the errors in knowledge and targeting accuracy?

How can the errors be reduced?

For the approach targeted aeroshells, what are the errors as a function of the deployment time?

Does onboard nav help and how much?

What is the dispersion due to passage of the aeroshell through the atmosphere?

Due to the time on the parachute?

Due to the error introduced by the small rocket firing?

Does Viking experience help in estimating the targeting accuracies?

What is the effect on the trajectory of the despin from 60 to 15 rpm following the targeting firing?

What is the effect of changing the assumptions or parameter values (e.g., flight path angle, ballistic coefficient, etc.)?

EI FOOTPRINT SENSITIVITY TO BURN ERRORS

(Approach Deployed Aeroshell)

- With nominal entry-interface (EI) conditions
 - $h_{EI} = 125 \text{ km}$
 - $\gamma_{EI} = -20 \text{ deg}$
 - $V_{\infty} = 3.5 \text{ km/s}$
- Accelerometer quantization = 0.005 m/s (typical)
- Assume ability to effect burn is comparable to ability to measure burn
- Thus, assuming 1 accelerometer quant of ΔV error per axis

DAYS PRIOR TO EI	1	2	5	10	20
$\Delta\gamma \text{ (deg)}$	0.01	0.02	0.06	0.12	0.24
$\Delta DR \text{ (km)}$	1.06	2.08	5.13	10.19	20.32
$\Delta CT \text{ (km)}$	0.28	0.55	1.34	2.68	5.34



MARS PENETRATOR VEHICLE ENTRY DISPERSION ANALYSIS

319

John P. Higgins
(617)-258-2433



ASSUMPTIONS

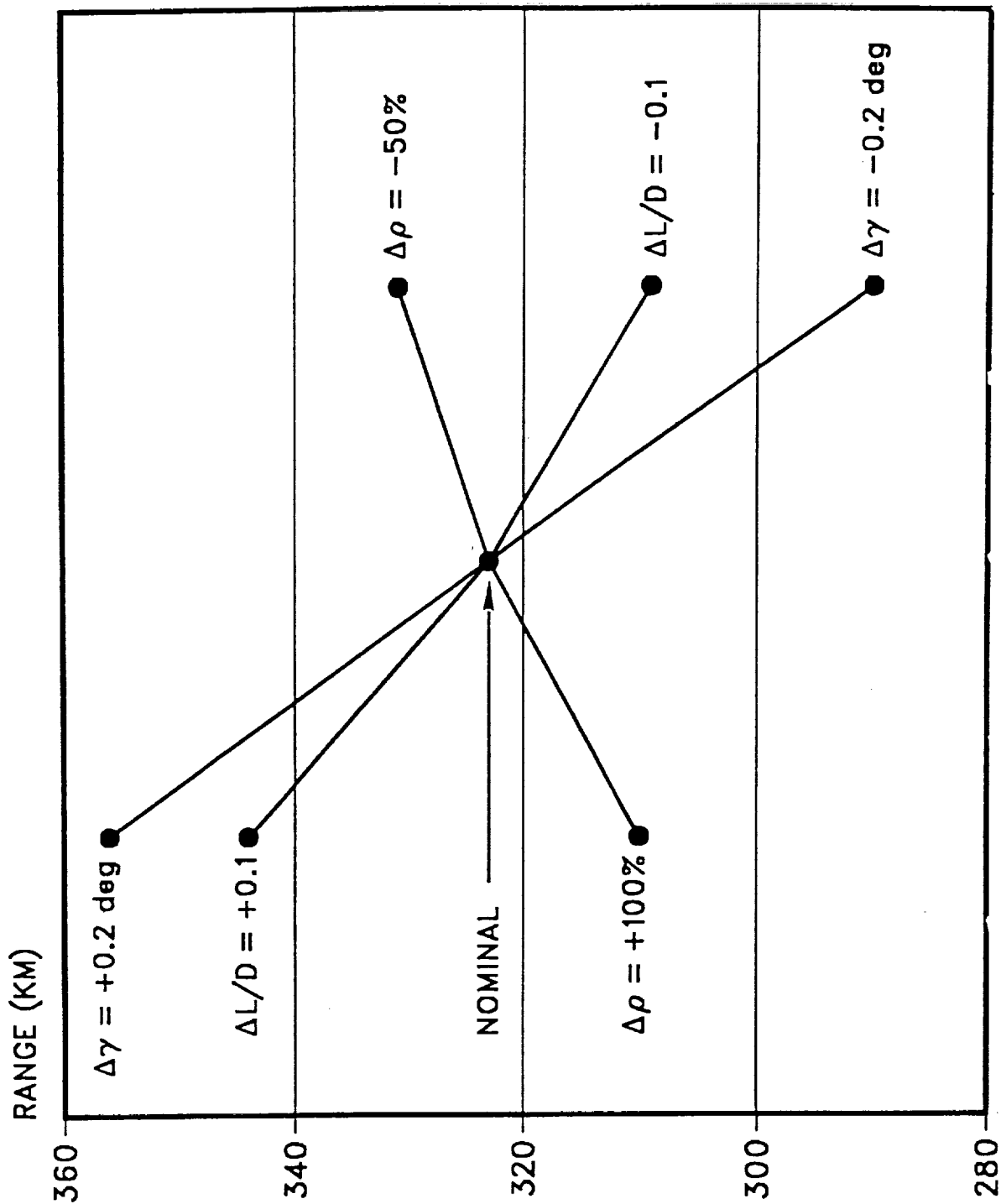
- PRE-DEORBIT TRAJECTORY
 - SEMI-MAJOR AXIS = 6986 KM
 - ECCENTRICITY = 0.485097
- ENTRY INTERFACE CONDITIONS
 - ALTITUDE = 125K KM
 - VELOCITY = 4.2 KM/S (DEORBIT PENETRATOR)
 - VELOCITY = 6.0 KM/S (DEPLOYED PENETRATOR)
 - GAMMA = -15° / -20° DEGREES
- BALLISTIC ENTRY (ZERO LIFT)
 - PENETRATOR BALLISTIC COEFFICIENT = 30 KG / SQ METERS
 - PARACHUTE BALLISTIC COEFFICIENT = 3 KG / SQ METERS
 - DEPLOYED AT 10 KM ALTITUDE
- ATMOSPHERE MODEL – MARS NORTHERN MEAN
- GRAVITY MODEL – CONIC



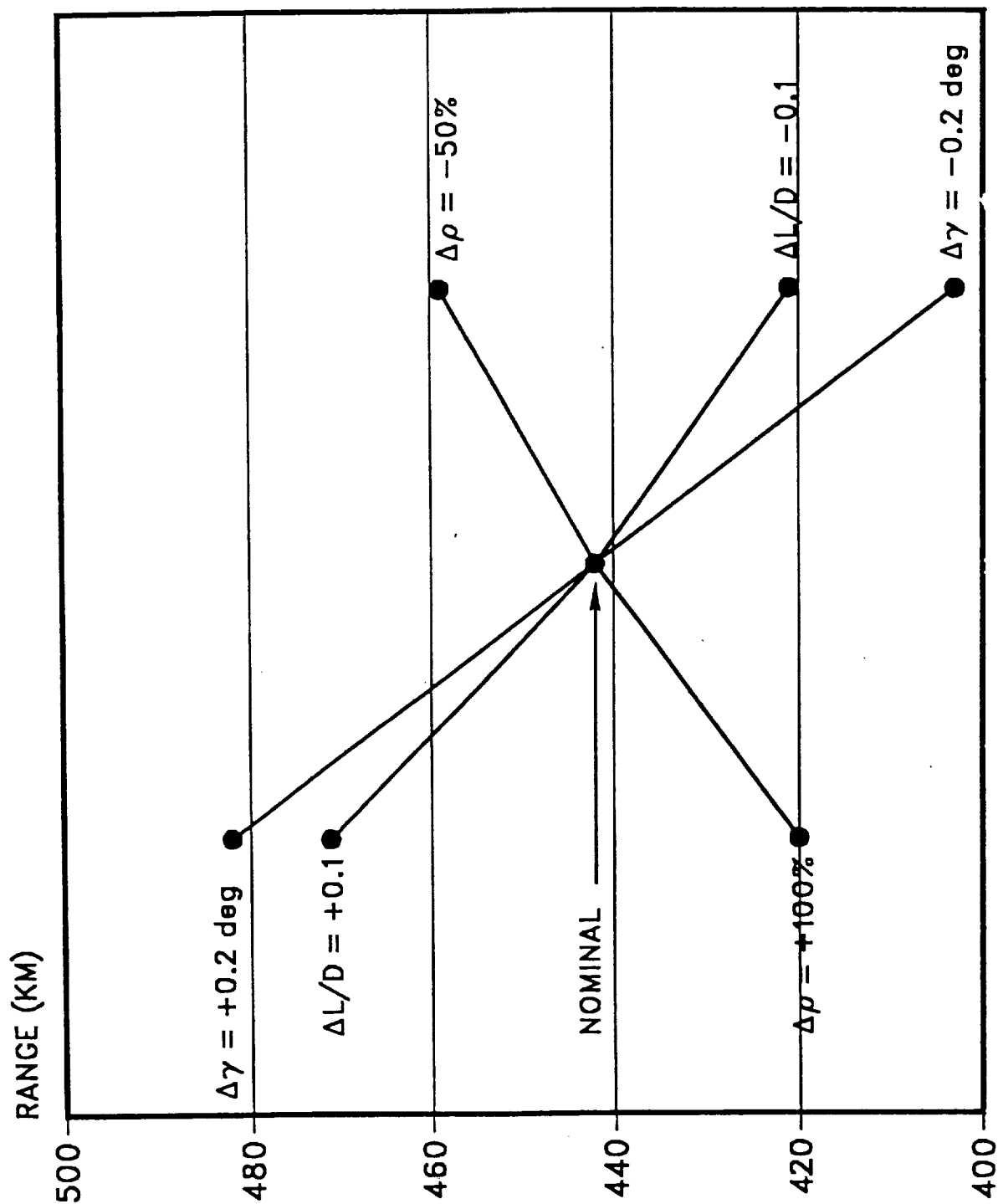
ENTRY DISPERSION SOURCES

- LIFT-TO-DRAG RATIO
- ATMOSPHERIC VARIATIONS
- ENTRY INTERFACE FLIGHT PATH ANGLE
 - DUE TO GN&C DEORBIT BURN PERFORMANCE (PRIMARILY NAVIGATION ERRORS)
 - DOWNRANGE DISPERSION PRIMARILY DUE TO ENTRY PENETRATION POINT VARIATION

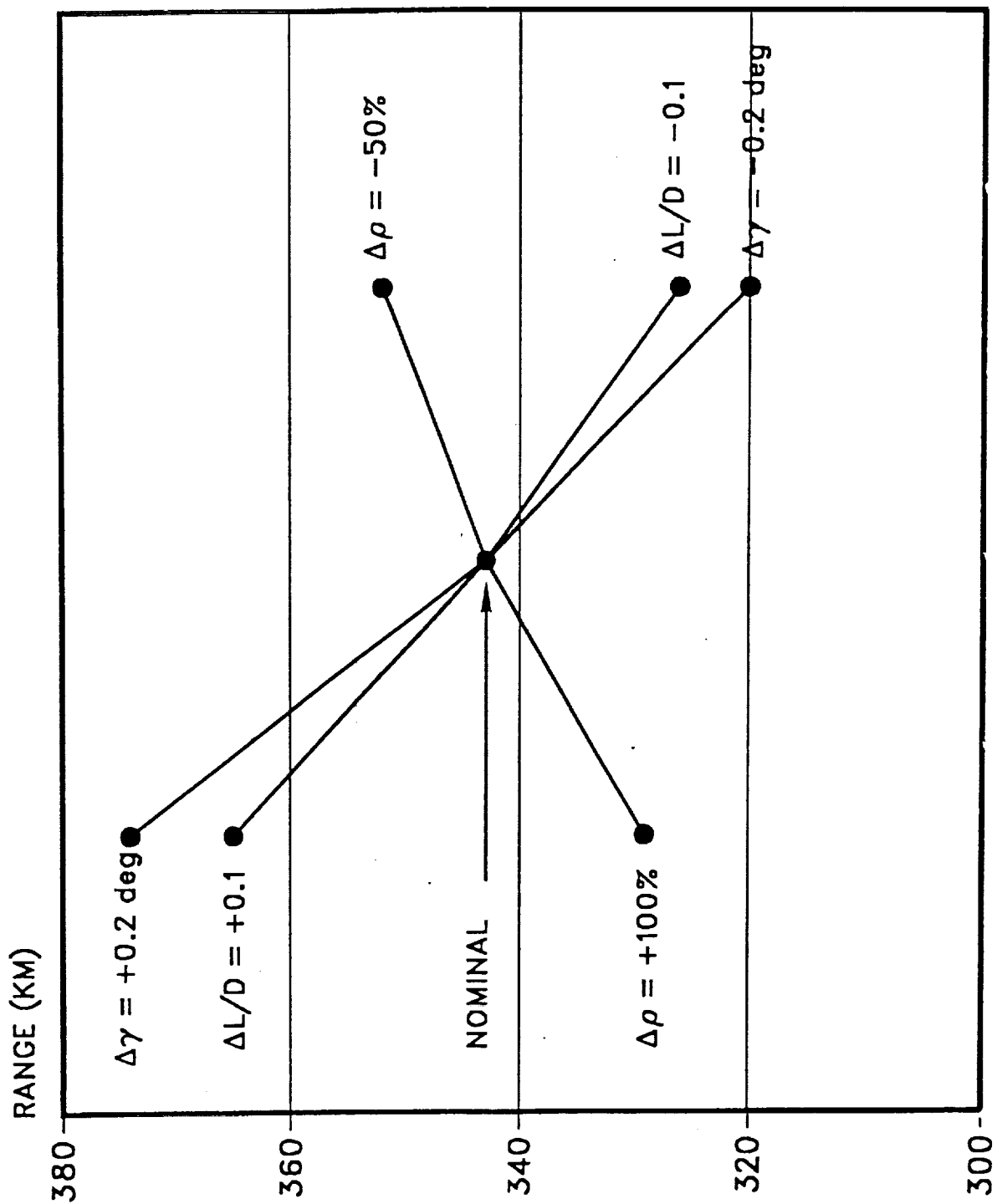
MARS ENTRY DISPERSION ANALYSIS DEORBIT PENETRATOR - EI GAMMA = -20 deg



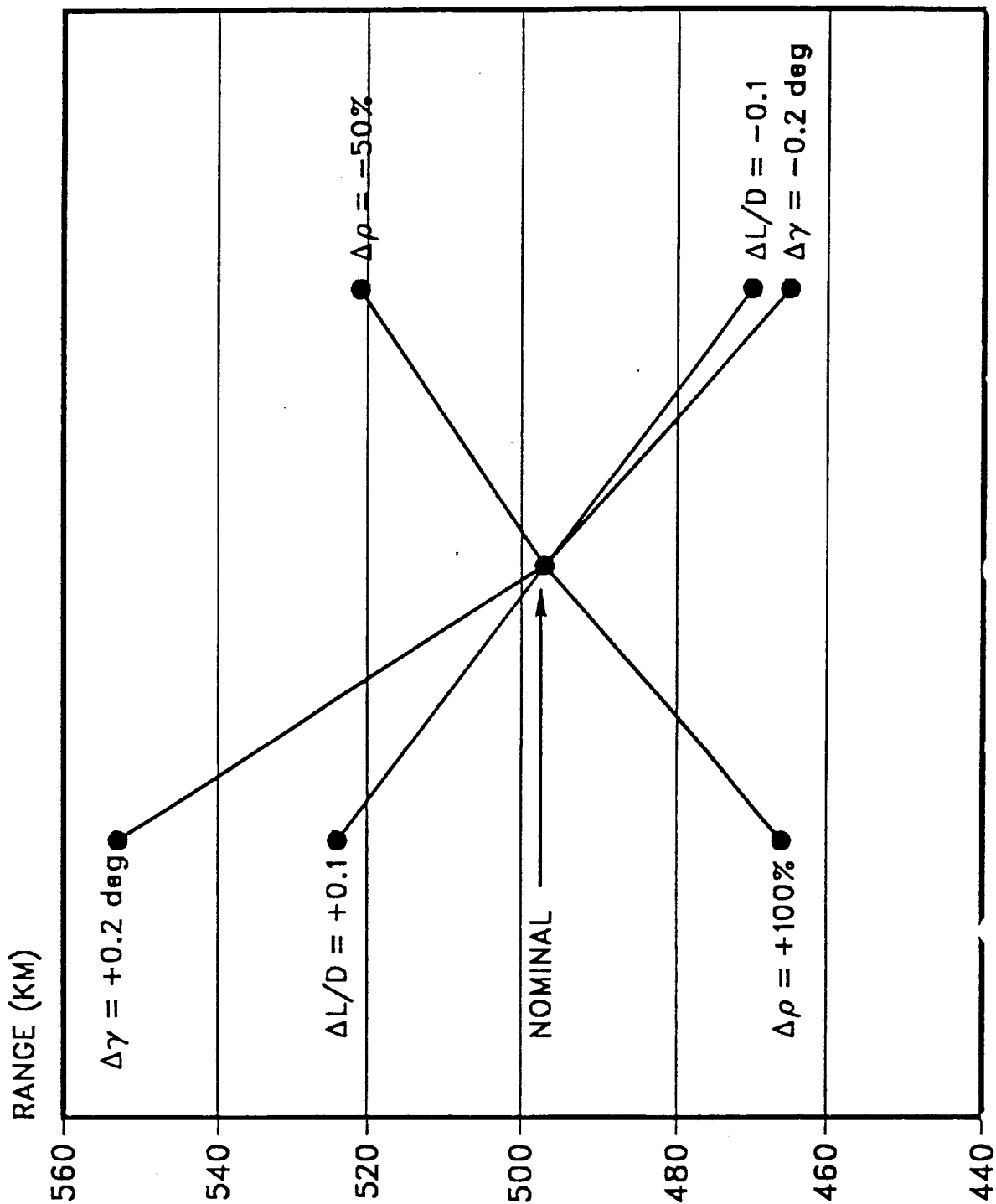
MARS ENTRY DISPERSION ANALYSIS DEORBIT PENETRATOR - EI GAMMA = -15 deg



MARS ENTRY DISPERSION ANALYSIS DEPLOYED PENETRATOR - EI GAMMA = -20 deg



MARS ENTRY DISPERSION ANALYSIS DEPLOYED PENETRATOR - EI GAMMA = -15 deg



Following is a list of memos produced at the Draper Laboratory on guidance, navigation, and control for the Mars Rover Sample Return Mission. Some of these may be relevant to the Global Network Mission studies.



**MRSR MEMO LOG
(MARS ROVER SAMPLE RETURN MISSION)**

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
1/3/89	001/EGB-89-001	Distribution	T. Brand	MRSR 1987-1988 Memo Log
1/18/89	002/EGB-89-022	Distribution	S. Bauer	Executive Summary of MRSR Lander Navigation Performance
1/12/89	003/EGB-89-023	Distribution	S. Shepperd	Summary - Mars Communication Orbiter as a Pre-Aerocapture Navigation Aid
2/7/89	004/EGB-89-034	Distribution	T. Brand	Mars Mission - Pre-aerocapture Navigation Communication Orbiter as a Nav Aid Preliminary Study
2/27/89	005/EGB-89-035	Distribution	S. Bauer/ T. Brand	MRSR Deorbit/Entry Navigation Performance with Updates from a Communications Orbiter
3/15/89	006/EGB-89-037	File	A. Engel	DSMAC Discussions in January at the Naval Avionics Center
4/7/89	007/EGB-89-084	File	D. Fuhry	High Energy Aerobraking Task Status
4/11/89	008/EJC-89-065R	T. Brand	R. Greenspan	Electronics Burden of Radio Navigation for a Mars Lander
5/11/89	009/EGC-89-118	Distribution	H.L. Malchow	Qualitative MRSR Requirements Survey
5/18/89	010/EGC-89-127	Distribution	L. Sackett	Flight Control Presentation at JSC on April 20, 1989

The Charles Stark Draper Laboratory, Inc.

MRSR MEMO LOG

DATE	MRSR-89-/EGB-	TO	FROM	SUBJECT
5/19/89	011/EGB-89-115	File	K. Spratlin/ T. Brand	MRSR Entry and Landing Task Presentation to MRSR Videocon
6/21/89	012/EGB-89-137	Distribution	A. Engel	Briefing Charts - Autonomous Landing Study Probability of Safe Landing
6/21/89	013/EGB-89-139	File	A. Engel	Briefing Charts - Safe Landings for MRSR (April 20, 1989)
	014 CANCELLED			
6/21/89	015/EGB-89-141	Distribution	D. Fuhry	GSDL Presentation Given at MRSR Mars Approach Navigation Coordination Meeting
6/21/89	016/EGB-89-142	D. Smith/JPL	D. Fuhry	Representative Computer Throughput Requirements for GSDL Predictor/Corrector Aerocapture Guidance
7/12/89	017/EGB-89-151	Distribution	S. Bauer	MRSR Deorbit/Entry Navigation Performance
7/31/89	018/EGB-89-169	Distribution	A. Engel/ S. Bauer	Briefing Charts - MRSR Probability of Landing and Deorbit/Entry Navigation Performance Studies
8/14/89	019/EGB-89-174	Distribution	T. Brand	Reduced IMU Alignment Error Prior to Deorbit and Entry
9/1/89	020/EGB-89-211	Dist.	H. Malchow	Attitude Control Configuration Options for the Trans-Mars Cruise Phase of the MRSR Mission Sample Return Segment

MRSR MEMO LOG

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
10/2/89	021/EJC-89-157	T. Brand	R. Greenspan	Power Considerations for Radio Navigation Aids during Mars Approach
10/3/89	022/EGB-89-216	K. Spratlin	W. McClain	Comments on "Trajectory Prediction Accuracy for Lower Order Mars Gravity Models"
10/3/89	023/EGC-89-225	Distribution	D. McCallum	Charts of Performance Specifications for Attitude Control Actuators and Sensors Rendezvous and Docking Sensors and Accelerometers
10/6/89	024/EGB-89-219	Distribution	S. Shepperd	Mars Approach Navigation Constant Power Radiometric Navigation Performance
10/6/89	025/EGB-89-221	Distribution	S. Bauer	MRSR Pointing Accuracy During Entry
10/26/89	026/EGB-89-235	Distribution	K. Spratlin	1989 MRSR Final Report Inputs
11/14/89	027/EGB-89-255	File	K. Spratlin	CSDL Contribution to Lunar/Mars Transportation Study Requested by D. Long/JSC
11/15/89	028/EGB-89-275	Distribution	D. McCallum	Attitude Control Requirements for Aerocapture Phases of MRSR Mission
11/15/89	029/EGB-89-260	Distribution	S. Bauer	Mars Entry Navigation Using IMU without External Updates
11/17/89	030/EGB-89-299	Distribution	S. Bauer	Mars Orbit Navigation with Radiometric Updates from a Communication Orbiter

MRSR MEMO LOG

DATE	MRSR-89-/EGB	TO	FROM	SUBJECT
12/31/89	031/EGB-89-303	File	D. Brown	Presentation on Sensor Look Angles During Hazard Avoidance

Session B, Submittal No. 12

Bruce A. Crandall
Hughes Aircraft Company

**LAUNCH VEHICLE ALTERNATIVE
STRATEGIES**

**BRUCE CRANDALL
HUGHES AIRCRAFT CO.**

LAUNCH/MISSION OPTIONS



SYSTEM	COST	SATISFACTION
BASELINE TIV/CENTAUR	100%	100%
2-3 ATLAS/CENTAUR *	~50%	≤ 100%
4-6 ATLAS/CENTAUR *	100%	>>100%

*EXISTING INTEGRATION FOR 6500+ LB CLASS, 8 FOOT BODY WITH
INTEGRAL PROPULSION

LARGE TRADE SPACE EXISTS DUE TO HIGH COST OF TIV/CENTAUR

PROBE INTEGRATION OPTIONS COMPARISON WITH PIONEER VENUS



	PROBE INTEG	INSERT PROP	ORBITAL SCIENCE	PROBE COMM	BUS CONTROL
--	----------------	----------------	--------------------	---------------	----------------

GNM CAR/ORB	X	X	X	X	X
-------------	---	---	---	---	---

PIONEER VENUS

MULTIPROBE	X			X	X
------------	---	--	--	---	---

ORBITER		X	X		X
---------	--	---	---	--	---

DIVIDING FUNCTIONS = LESS COMPLEX INTEGRATION
MORE ORBITAL FLEXIBILITY
DEPLOYMENT
COMMUNICATIONS

SUMMARY



MAINTAINING A BROAD VIEW OF THE GNM OPTIONS AT THIS TIME
WILL LEAD TOWARD THE MOST OPTIMUM DESIGN

SIGNIFICANT MATURE TECHNOLOGIES EXIST IN THE AREA OF HIGH G
ELECTRONICS, PROPULSION, IMAGING AND GUIDANCE...WITH DIRECT
APPLICABILITY TO..

PENETRATOR DESIGN
DEPLOYMENT PHILOSOPHY
CARRIER INTEGRATION
TARGETING FIDELITY

BEST SATISFACTION TO COST PERFORMANCE MAY BE REALIZED
USING SMALLER LAUNCH VEHICLES AND SATELLITES

SCIENCE MISSION PRIORITIZATION AND CHARACTERIZATION WILL
HELP TO CONVERGE MISSION REQUIREMENTS AND DESIGN

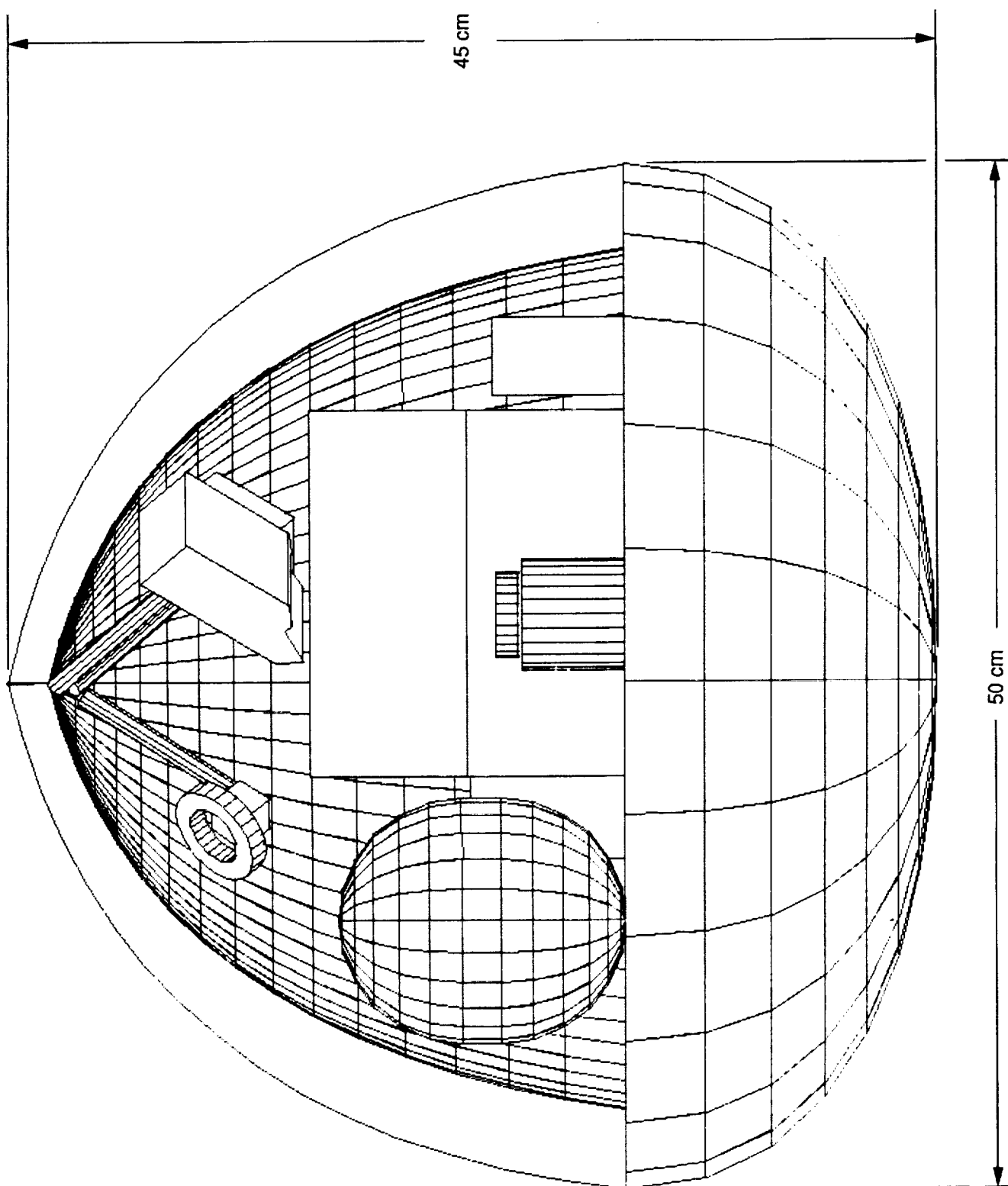
Session B, Submittal No. 13

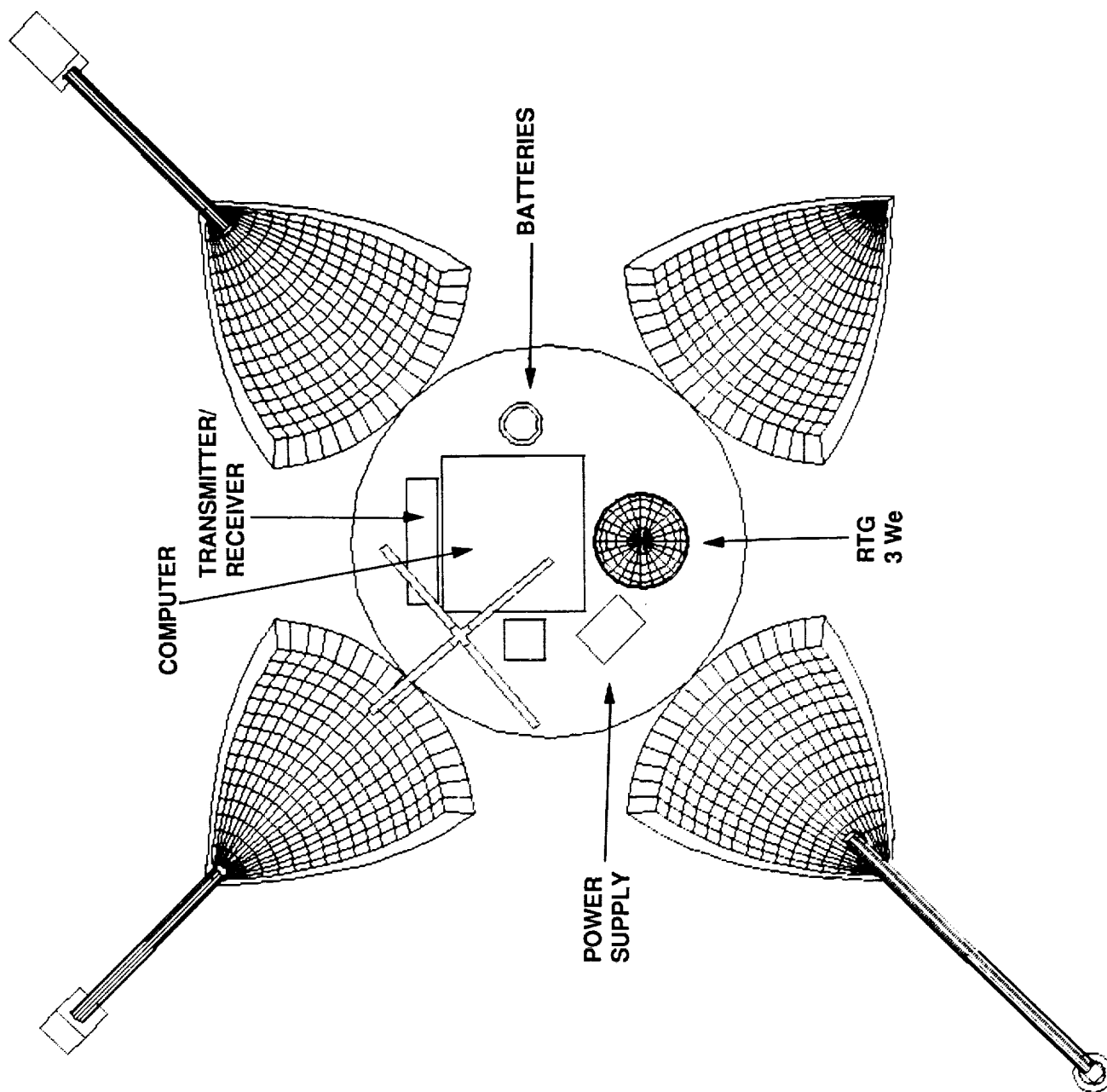
Carlos S. Moreno
Jet Propulsion Laboratory/California Institute of Technology

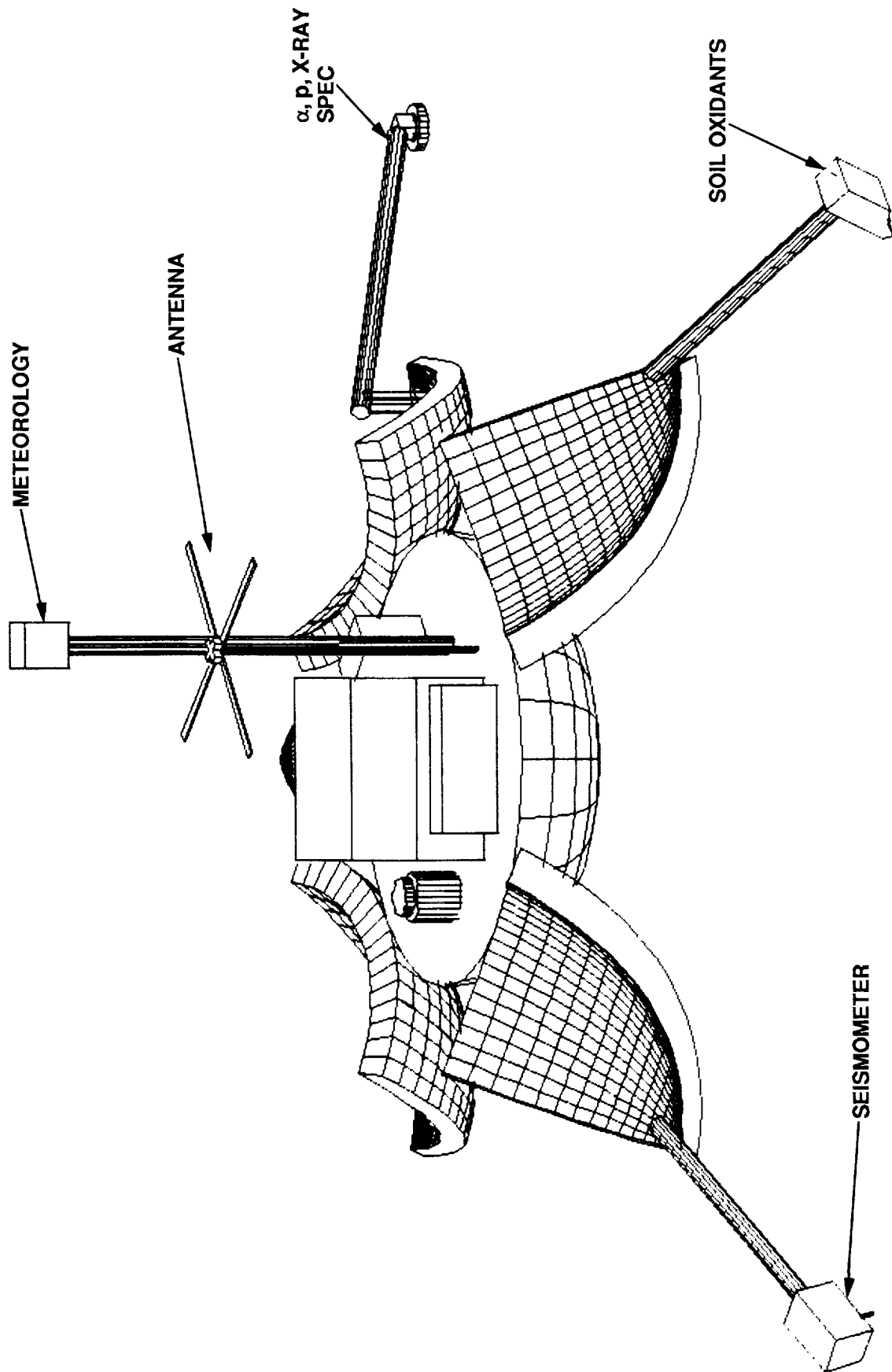
HARD LANDER CONCEPTUAL DESIGN

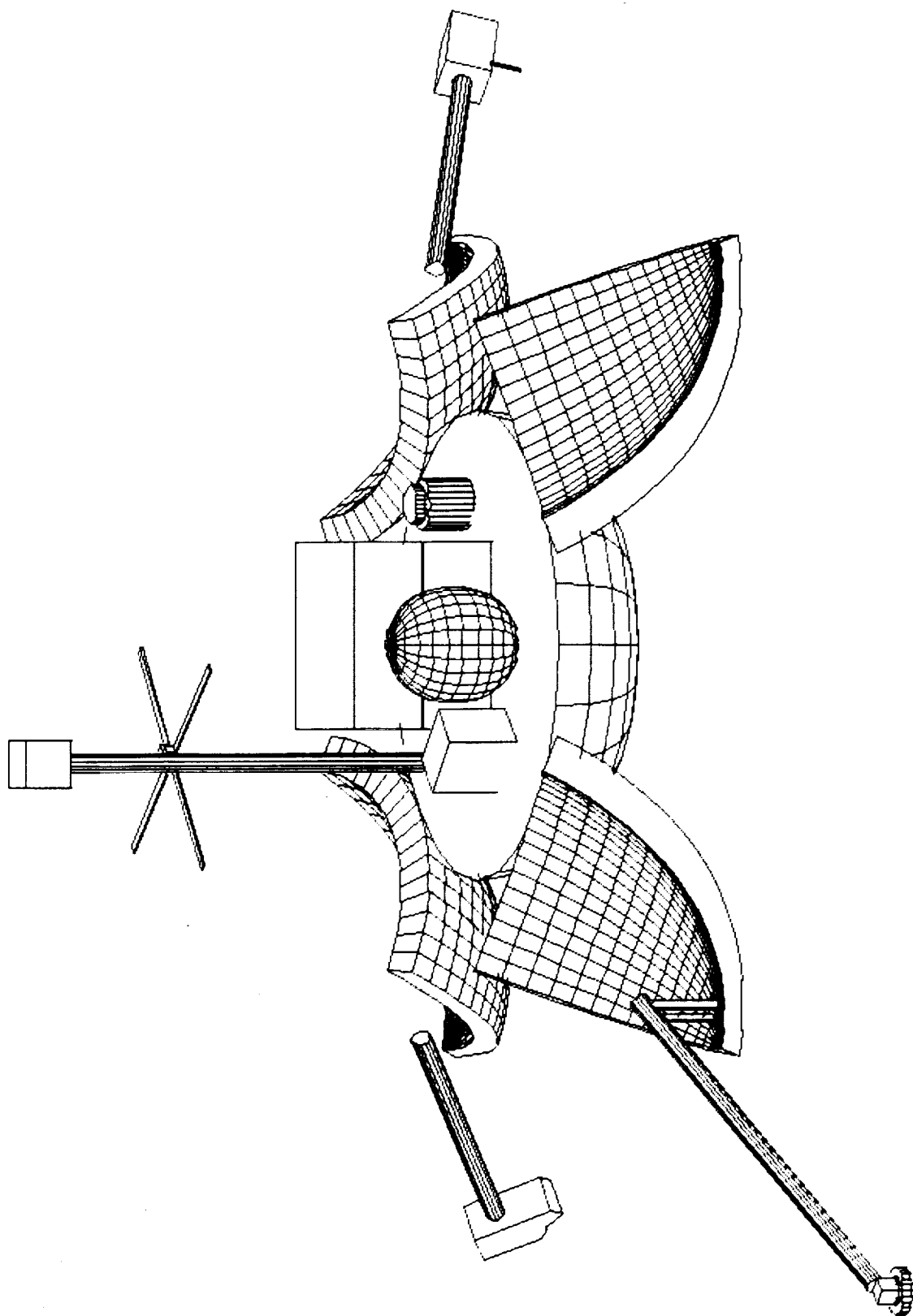
CARLOS MORENO

JPL

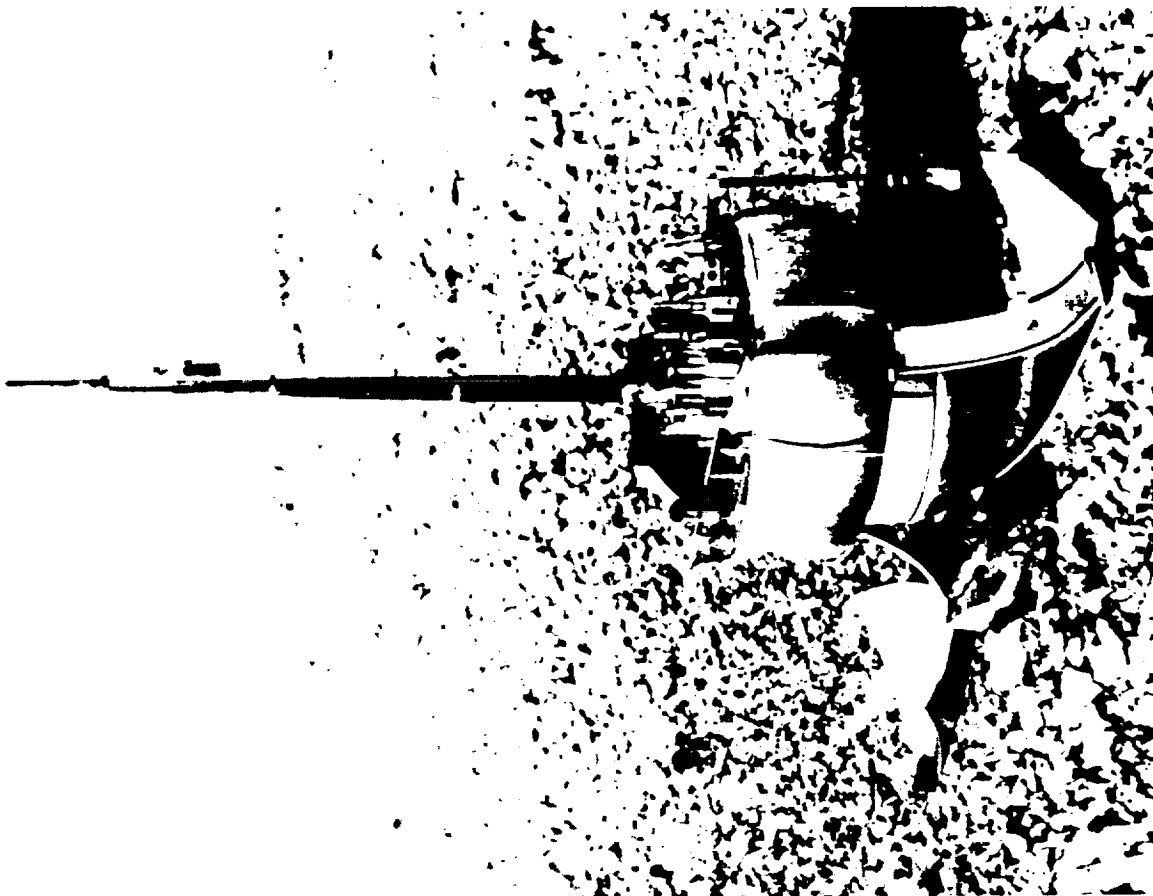








An automated biological laboratory was developed by Philco Aeronutronic Company to study the possibility of a hard-landing entry probe to make simple assays of the Martian environment. One of several studies for Voyager in the mid-1960s, it grouped science instruments that could be programmed for numerous experiments. None of the projects was flown, but they provided understanding of extraterrestrial biology detectors.



Session B, Submittal No. 14

John Garvey
McDonnell Douglas

DELTA LAUNCH VEHICLE STRATEGIES

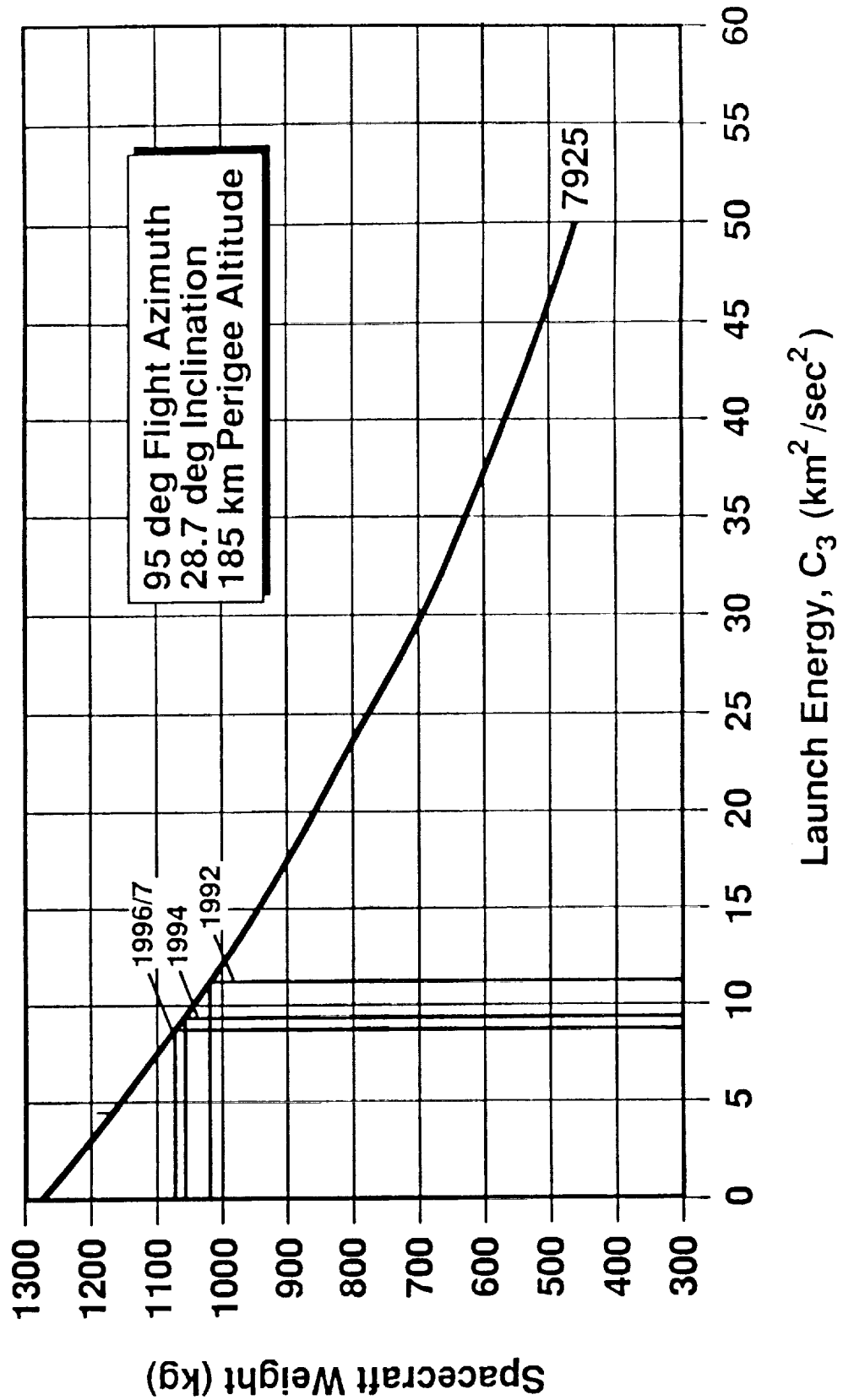
JOHN GARVEY

MDAC

DELTA THREE-STAGE PLANETARY CAPABILITY – ESMC

VJX098.1 T1CV

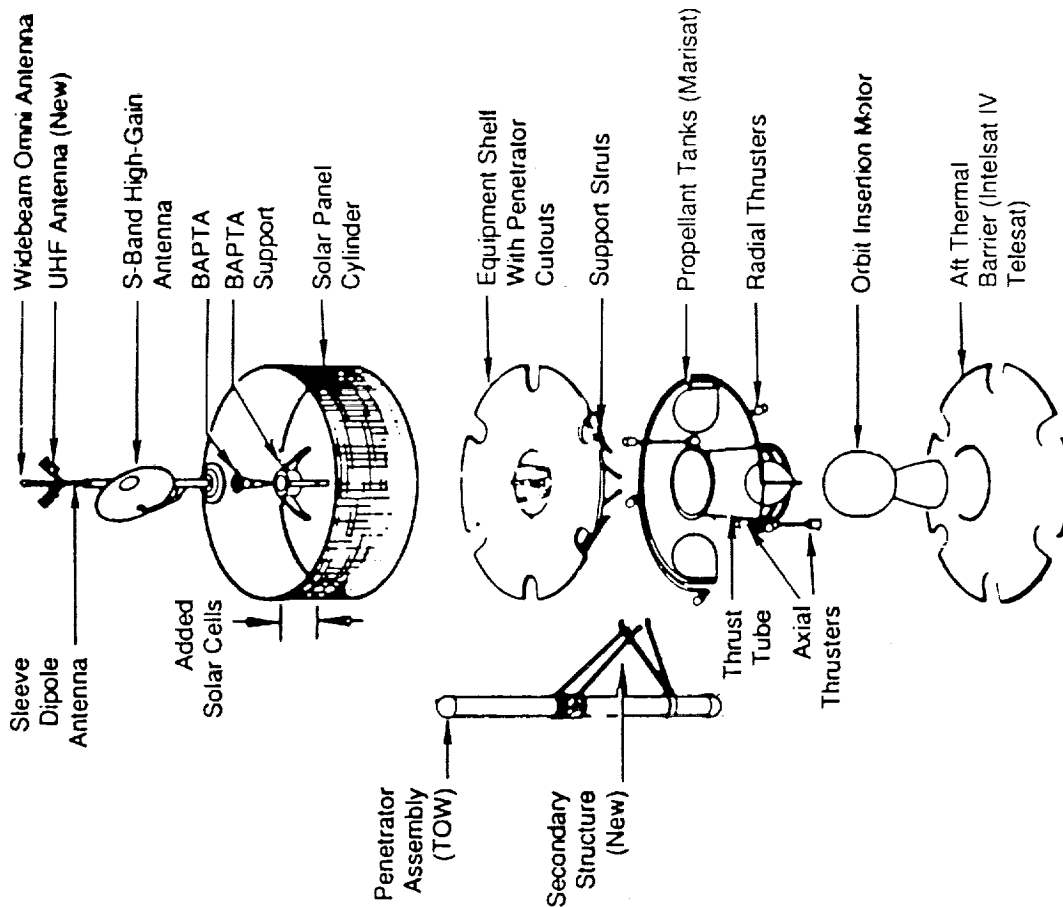
MCDONNELL DOUGLAS



MSP SPACECRAFT BASED ON PIONEER-VENUS BUS

VJX 144 T2CA

MCDONNELL DOUGLAS

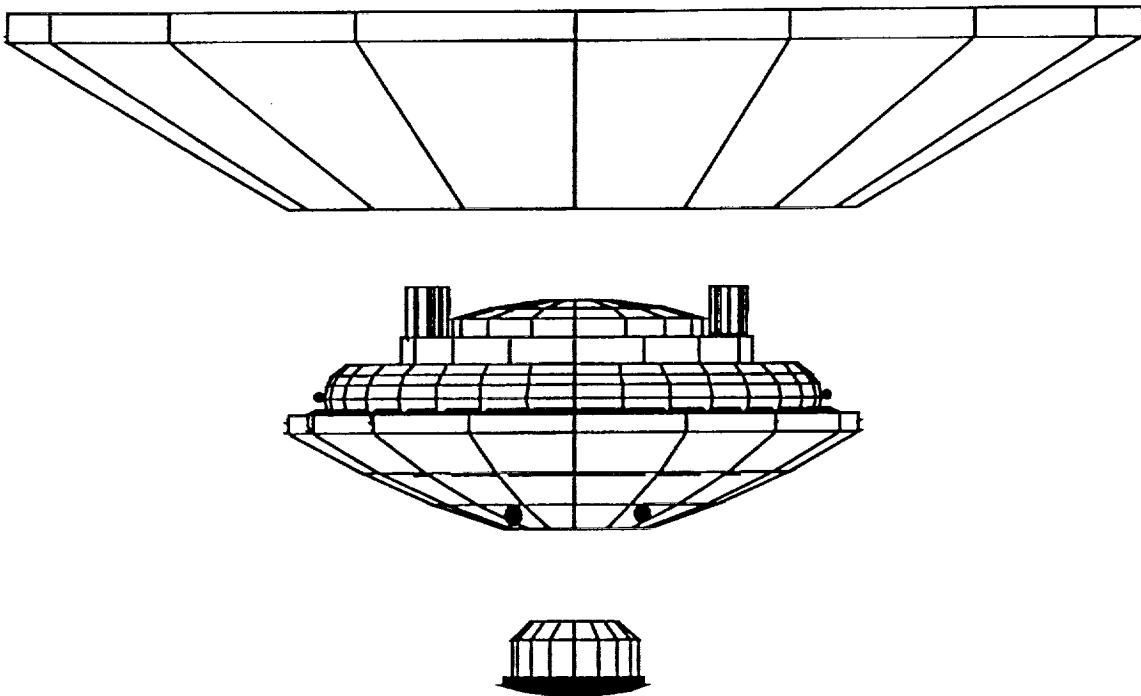


Session B, Submittal No. 15

Stephen Bailey
Johnson Space Center

GNM Mission and System Design Proposal

A Position Paper in Response to the Mars Global
Network Workshop, Feb 6-7, 1990



Stephen Bailey
NASA/JSC/IZ3
NASAMail SBailey
(713) 283-5411

A GNM Mission and System Design Proposal

A Position Paper in Response to the Session B of the Mars Global Network Workshop, Feb 6-7, 1990

Introduction

After having attended the Mars Global Network Mission (GNM) Workshop, and upon some reflection, I have put together a mission and system design option for the GNM which I believe is complementary to the 2001 Sample Return Mission (SRM). In this paper I take an advocacy position for the proposed mission; it is not intended to be an objective review, although both pros and cons are presented in summary. This work represents my own opinions and judgements, and is not an SRM policy statement, nor is it supported by any systematic analysis. These ideas are an expansion and elaboration of the design proposed by Al Friedlander of SAIC in the Session B discussion of the GNM workshop.

In arriving at the proposed design I used the following criteria, in order of priority, for evaluation:

- 1) Maximize Science Value
- 2) Keep Costs Low
- 3) Maximize Heritage (both from previous missions and heritage to be provided to future exploration missions, particularly the SRM)
- 4) Design to fly in the earliest possible opportunity
- 5) Make it "Innovative"

The Elements of the proposed mission are:

- 1) Aeroshelled Landers
- 2) Communication Orbiter(s)

Mission Scenario

The mission consists of launches from earth in the '96, '98, and '01 opportunities on Delta-class launch vehicles (~1000 Kg injected to Mars in 8 to 10 ft diameter shroud). The trans Mars boost stage injects a stack of small independent, aeroshelled spacecraft. The stack separates from the boost stage and each rigid (as opposed to deployable) aeroshell flies to Mars on its own, performing midcourse maneuvers as necessary. On-board GN&C systems provide precision pointing (via torque wheels) and burn execution. Each spacecraft flies a unique trajectory which is targeted to achieve approach atmospheric interface at the desired latitude and lighting conditions; arrival times may vary by a month or more. A direct entry is performed, there is no propulsive orbit capture. The aeroshelled rough-landers are targeted to achieve a desired attitude and entry flight path angle, and then follow a passive ballistic trajectory until terminal descent. Based on sensed acceleration (integrated to deduce altitude), the aft aeroshell skirt is jettisoned, a short time later a supersonic parachute is deployed. The ballistic coefficient

of the parachute is sized to achieve terminal velocity at about 8 Km. However the parachute is not deployed until a few Km above the surface to minimize wind-induced drift. This relatively short period on the parachute is possible because of the low ballistic coefficient of the aeroshell, and allows surface sites up to 6 Km above the mean surface level to be visited. The nose cap (weighted by the no longer required torque wheel assembly) is jettisoned and descent imaging begins, a laser altimeter also measures true altitude. (Depending on what altitude descent imaging is first required, the nose cap may be jettisoned prior to the aeroshell skirt jettison.) Based on range and range rate to the surface, the parachute is jettisoned and the lander uses descent engines to achieve touchdown velocity. (Note: if the ballistic coefficient of the aeroshell is sufficiently low, a parachute is not required, the ballistic terminal velocity provided by the aeroshell would be low enough that a propulsive descent could be performed directly). A contact sensor shuts down the motors to avoid cratering, and the lander rough-lands at less than 5 m/sec. The remaining aeroshell and a deployable bladder attenuate landing loads and minimize the possibility of tip over. Science instruments are deployed and activated, and the network is established.

See the appendix of figures which illustrate the mission and spacecraft designs.

Shared Communications Infrastructure

In this scenario, the communications relay orbiter(s) are provided as infrastructure for both the GNM and the SRM. In the reference GNM and SRM scenarios, each mission provides its own communications system. These systems are a part of the carriers which are captured into deployment and (in the case of the SRM) retrieval orbits; these orbits are not the preferred ones from a communications standpoint, and may in fact be far from optimum. Because of the successive nature of these missions, commonality between the communications system requirements should be explored. Because of the stated commitment to planetary exploration, consideration should include the use of this system to backup or augment future, higher capability Mars communications systems.

Deployment from the Trans Mars Boost Stage Contrasted to the Reference GNM Mission Scenario

Another key feature of this design proposal is the lack of a centralized carrier vehicle which propulsively captures into Mars orbit and performs deployment of landers from that orbit. In the proposed approach, the aeroshells are separated for the boost structure via a simple sequencer. They then become independent spacecraft, each targeted and tracked on a unique trajectory.

In contrast, the reference GNM mission designs involve a combination of deployment from orbit and deployment on approach.

Although an orbit design exists which satisfies lighting conditions over a wide range of latitudes, including polar (re. "A Polar Orbit Mission for the Mars Global Network Mission", Philip Knocke, JPL), it comes at some expense. The 1/5 sol polar orbit requires a higher capture Delta-V than a more elliptic orbit.

A 160 day wait (for the 1998 launch opportunity) is also required to achieve the correct orbital conditions before entry vehicle deployment may begin. The opportunity available then sweeps from the south pole to the north over a 180 day period - thus to get full latitude coverage and emplace the full network would take almost a year from the time of arrival at Mars.

The carrier deployment strategies discussed in the workshop considered the deployment of aeroshells with no active GN&C system. In this scenario, the carrier would provide pre-deployment pointing and would deploy the aeroshell in such a way that tip off rates were negligible; the aeroshell would then simply execute a fixed delta-V burn. This "point-and-shoot" strategy for aeroshell deployment on approach variety has the virtue of simplicity, but at the expense of landing accuracy (especially for low entry flight path angles). Of course this accuracy can be improved by putting a GN&C system on the aeroshell. Also execution accuracy for approach can be improved by a combination of steep entry flight path angle and simply delaying approach deployment until the last "minute" (2 days out, 1 day out, hours...?). Waiting however, incurs a Delta-V penalty.

The design choice of putting an on-board GN&C system then leads one to the scenario proposed here. That is to deploy aeroshells on approach, but that deployment may begin immediately after the Trans Mars Injection (TMI) burn. In this way the aeroshells are independently guided to entry interface from post TMI separation from the boost stage. Since the aeroshells all perform direct entry they are all of the same design (ie. there are no disparities between having to design both orbital deployed and approach deployed aeroshells).

The Development of a Spacecraft Bus

Whether or not an existing bus such as the Mars Observer bus can be used, there is significant development, integration, testing and certification to go through prior to launch. A closer look is warranted to compare the costs of developing a large central bus with separate aeroshells as contrasted to developing a simple aeroshell deployment mechanism and many small, independent spacecraft. The development of many smaller and simpler appears to have great potential to lower the overall costs of the GNM mission, and may help moderate costs for the following sample return mission by providing valuable infrastructure and heritage for the SRM program.

This leverage would be provided by the design of a single small aeroshelled lander which could have broader application than the currently proposed penetrator concept. Once a kick stage has provided the necessary trans-Mars Delta-V, only attitude maintenance and periodic midcourse corrections up to the point of entry interface are required for the proposed spacecraft. The GN&C heritage to solving this problem is vast, and an off-the-shelf solution requiring little more than integration is possible, given the current trend towards miniature satellites. A spacecraft required to do orbital insertion and orbital deployment is in my opinion an unnecessary complication. Each aeroshell would simply maintain course and attitude until entry interface, and from there follow a passive ballistic trajectory (no aeromaneuvering) up to terminal descent.

Mission Strategy

There is a possibility that a vigorous, aggressive development schedule could produce a '96 launch. This is possible because of the strong heritage that exists from previous and current engineering and development efforts. In any case, the science objectives and program enhancing opportunities available from this proposal argue for launch in multiple opportunities. For instance, if the scheduled launch dates were in successive years (say '96, '98, and '01), a unique strategy for mission reliability exists. If a first attempt at an attractive site fails, PLAN on trying again later instead of sacrificing global placement for a strategy of sending two landers to every site in order to achieve redundancy. Or, if every thing works on the first try and the network is satisfactorily established - stop, you're finished, no extra launch or set of launches is required.

The Advantage of Smaller Independent Spacecraft

The idea of simple independent carriers has a number of other advantages:

- 1) It allows smaller, simpler launch vehicles like the Delta or Atlas to be used (while still allowing the launch of a GREATER number of landers from a Titan IV than currently planned), which translates both into costs savings for the agency and much greater launch flexibility.
- 2) The mission is adaptable at modest cost. The global network can be sustained, added to, or evolved incrementally as questions arise, objectives evolve, and instrumentation improves.
- 3) The payload bay is reconfigurable (more so as compared to a penetrator fore/aft body design). The science equipment bay on the proposed lander is reconfigurable to accommodate 20 Kg of science instruments specific to latitudes or science objectives.
- 4) The design is reusable and provides heritage to the SRM. There may be tremendous design leverage to be found in the SRM if the sites selected for the SRM can be visited by simple landers (either carried piggy back and deployed on approach or launched separately), that provide exact terrain knowledge at the site and establish navigation aids that lead the lander to a landing area verified to be safe per lander design. Using GNM heritage, this could be done at a fraction of the cost of a comparable imaging orbiter mission. The Human Exploration Vehicles could use these "throwaway" landers in a similar fashion, and to conduct specific surface experiments related to site selection.
- 5) The design may be suitable for micro-rover ("Ant") deployment.
- 6) The aeroshells may be placed with relatively high accuracy by employing radiometric approach navigation via the communications orbiter(s). This would provide a flight demonstration for this navigation technique for the SRM while enhancing the GNM. A high factor of safety for the GNM is retained since earth based navigation would probably be the primary method.
- 7) Engineering heritage for future possible missions. A modified aeroshell bus (without the aeroshell skirt) could be used as a flying testbed for various L/D configurations by modifying the aft aeroshell skirt. The testbed could be used to evaluate various GN&C algorithms and would as a bonus extend our operational understanding of the variabilities of the Martian atmosphere. This kind of testbed may be the most cost effective method of getting operational aerocapture experience at Mars. The aeroshell bus will also fit inside very small launchers such as the Orbital Sciences Pegasus or Taurus, or the General Dynamics Atlas-E. A deployable aeroshell skirt could be developed (which could have a much lower achievable ballistic coefficient), with a modified bus used in flight test and operations. This has the additional advantage of sending a large number of probes through the martian atmosphere thus building the engineering knowledge database of Mars atmospheric flight prior to launch of a Human exploration mission.

Technology

I believe this proposal can be accomplished with minimal technology risk. This may not qualify it as "technologically innovative", but I see no need to invent technology where it obstructs timely, cost effective execution of the mission. The possibility of pressing for a '96 launch should be investigated. However, for serious consideration of a '96 launch, funding for concept studies needs to be provided now.

For the mission proposed here, the program risk that I believe exists for early launch of high G designs is mitigated. This is a simple mission, with a single simple spacecraft to design (excluding the comm orbiter which has even greater heritage working for it). I am sure that no show stoppers exist for penetrators, but there seem to be significant development costs and schedule risks associated with them. The fact is that none of the instruments, with rare exception, have been developed and tested for the very high G environments, and I am not aware that INTEGRATION of this number and variety of high G instruments has ever before been attempted (CRAF penetrator is the nearest data point that I am aware of, but the G loads there are considerably lower than those considered for the Mars penetrators, especially the aft body G loads). The combination of designing for the intense thermal flux, radiation, and G load environments, have probably not been predominate considerations for the majority of past high G development programs.

In this proposal, the strategy was to provide a relatively generous 20 Kg science payload capability with an ample 10 watt constant power supply augmented with rechargeable batteries. Several types of science payloads can be envisioned, each tailored of objectives which vary with latitude and the required number of a particular experiment type. As far as satisfying the requirements which lead to penetrator designs (subsurface sampling, placement of seismic geophones) a number of proposals emerged in the Session B workshop for satisfying these requirements. For instance a flexible, cable driven drill for acquiring subsurface regolith samples to a depth of up to 3 meters should be quite possible to incorporate into one such payload type. Geophones may be placed away from the lander on teathers to reduce the chance of interference, or they may be driven into the surface with a pyrotechnic device. I believe that the consensus at the meeting was clearly that engineering solutions could be found to satisfy science objectives, whether the surface device was a soft, rough, or hard lander. For the proposed rough lander design, risk and cost are mitigated by the using current expertise in developing, integrating, and testing moderate G instruments (10's of G's instead of 100's or 1000's).

Mass Guess-timates

Subsystem or Component	Mass (Kg)
Science Payload (including atmosphere profiling)	20
Structure (primary and secondary)	45
Power	
RTG's (2)	5
Batteries	5
Communications	10
GN&C/Propulsion	
---Avionics	10
---Torque Wheel Assembly	10
---Attitude Control System (Spin/Despin) and RCS Hardware	10
---Tanks & Fuel	35
Thermal Protection	30
Thermal Control	5
Parachute Assembly	15
Miscellaneous	5
Total	205

Notes:

- 1) This breakdown was used to get a rough estimate of the total mass. The numbers here represent only an educated guess, actual mass may vary, perhaps significantly, from these based on a detailed requirements analysis of the Global Network mission, and a comprehensive mass assessment.
- 2) For an aeroshell diameter of 2.44 m (8 ft) the ballistic coefficient would be about 44 Kg/m², for a diameter of 3.05 m (10 ft), all other things being equal, the ballistic coefficient is about 28 Kg/m². Lower ballistic coefficient translates into higher entry G-loads and heating rates, but also into steeper achievable entry flight angles which improve landing accuracy and provide the ability to achieve higher (polar) latitudes; lower ballistic coefficient also means lower mach numbers, or subsonic conditions, at parachute deployment. Exactly what latitudes are achievable should be the subject of future study.
- 3) Usable payload volume is about 50 cubic centimeters (1.8 cubic feet).

Challenges...

This proposed design is certainly not without technical and programmatic challenge. I encourage others to critique this proposal, but some difficulties I can see with the design are:

- 1) Navigating a fleet of vehicles to Mars simultaneously. This may saturate an already oversubscribed DSN. An integrated DSN upgrade or an alternate communications and navigation approach or system may have to be pursued.
- 2) A systematic injected mass study may show that some of my estimates are significantly in error. For instance, a stellar sensor such as the Ball CS-203 is required to provide inertial attitude reference, but even the CS-203 at 5.5 Kg, 6 watts, and 9 arcsec accuracy is not as small or precise as desired; a lightweight, low power Canopus tracker is assumed to be available. A total target weight of less than 200 Kg is attractive, I believe that lower weight (and thus lower ballistic coefficient, higher achievable latitudes, and higher landing accuracy) is attainable given the current trend towards micro-spacecraft. In any case, using off the shelf miniaturized components and technology is key to the success of the proposed design approach. (Is this technologically innovative?)
- 3) Science objectives best accomplished from orbit will require another orbiter (Son of Mars Observer?), or perhaps science payloads could be piggy backed on the (separate) communications orbiter.
- 4) Establishing a shared communications infrastructure may be a challenge. The communications and operations requirements of the missions need to be analysed together to determine what the best approach is to solving both problems. The placement of GNM landers at the poles, for instance implies the need for highly inclined relay orbits, while a sample return operation may best be satisfied with an aerosynchronous relay orbit.
- 5) Achieving the desired (steep) entry flight path angles from approach velocity may be problematic. Heating rates and total heat load are of special concern. The proposed approach would rely heavily on the heritage of Shuttle, AFE, and the High Energy Aerobrake work currently underway for Thermal protection materials, heat resistant substructure, and insulation materials and techniques.
- 6) Mission planning to achieve the desired distribution of landers at preselected longitudes and latitudes at the proper lighting conditions for descent imaging may be constrained by orbital mechanics and the launch dates, combined with the achievable entry flight path angles (function of ballistic coefficient, G loads, heating - requires further analysis). It may be necessary to relax the lighting condition requirement for descent imaging for some of the sites.
- 7) Achieving a '96 launch date would require an immediate commitment to GNM concept studies, and an innovative approach to contracting, developing, managing, and administering the program.

Conclusions

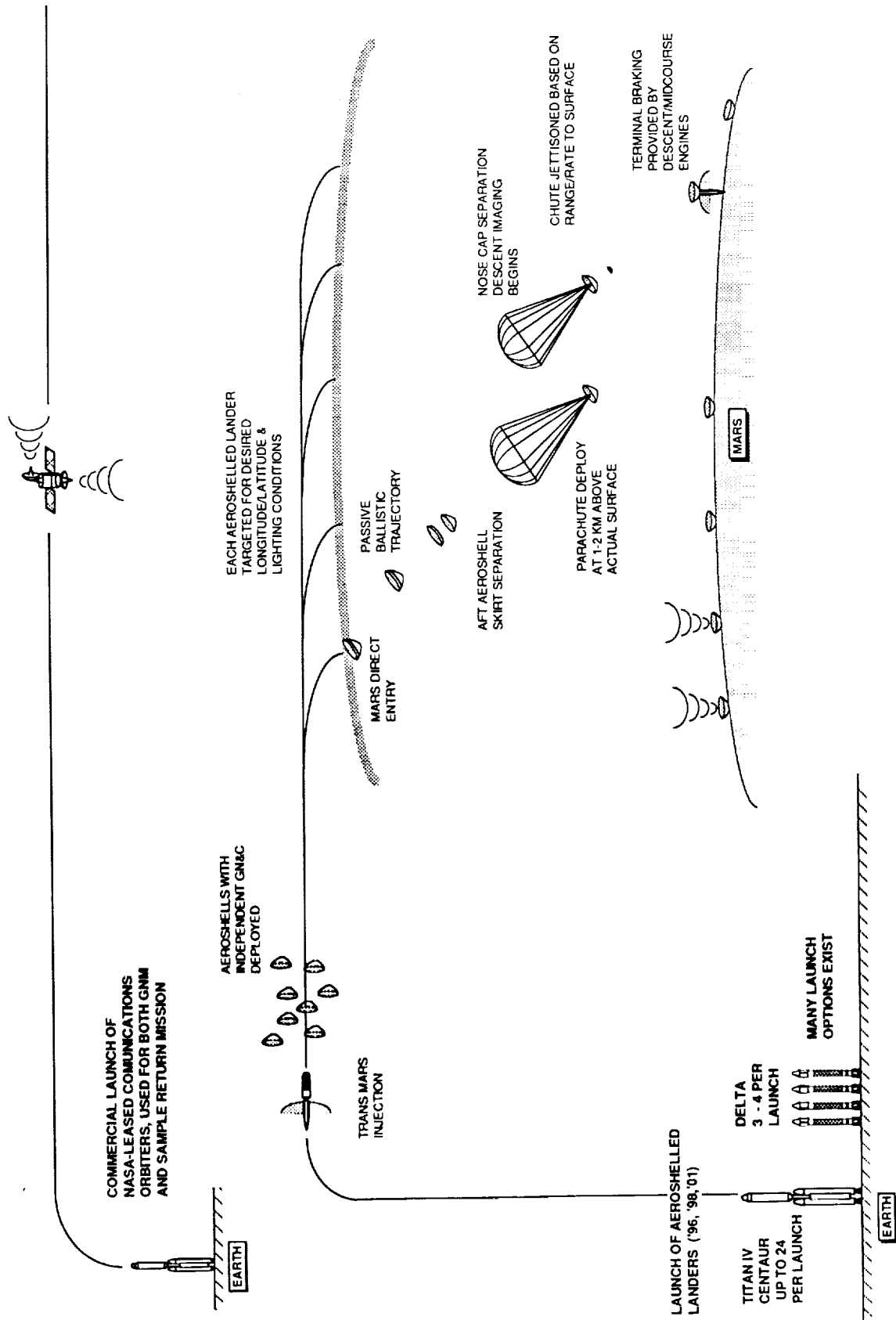
An approach such as the one proposed by Al Friedlander, which I have elaborated on here, has great promise in terms of reduction of cost and risk, increased flexibility, heritage and commonality, and I believe can reap substantial political dividends as well. However, a system engineering and cost estimation effort is needed to ascertain what the payoff of such a proposal might be. For a serious investigation of the possibility of a '96 launch of any description, it is imperative that funding of these important concept studies be swiftly provided.

While there is much refinement and analysis needed for this proposal, it has attributes which I hope will receive serious attention. My hope is that this and other proposals can generate the kind of discussions which will lead to a well balanced Robotic Exploration Program and Human Exploration Initiative.

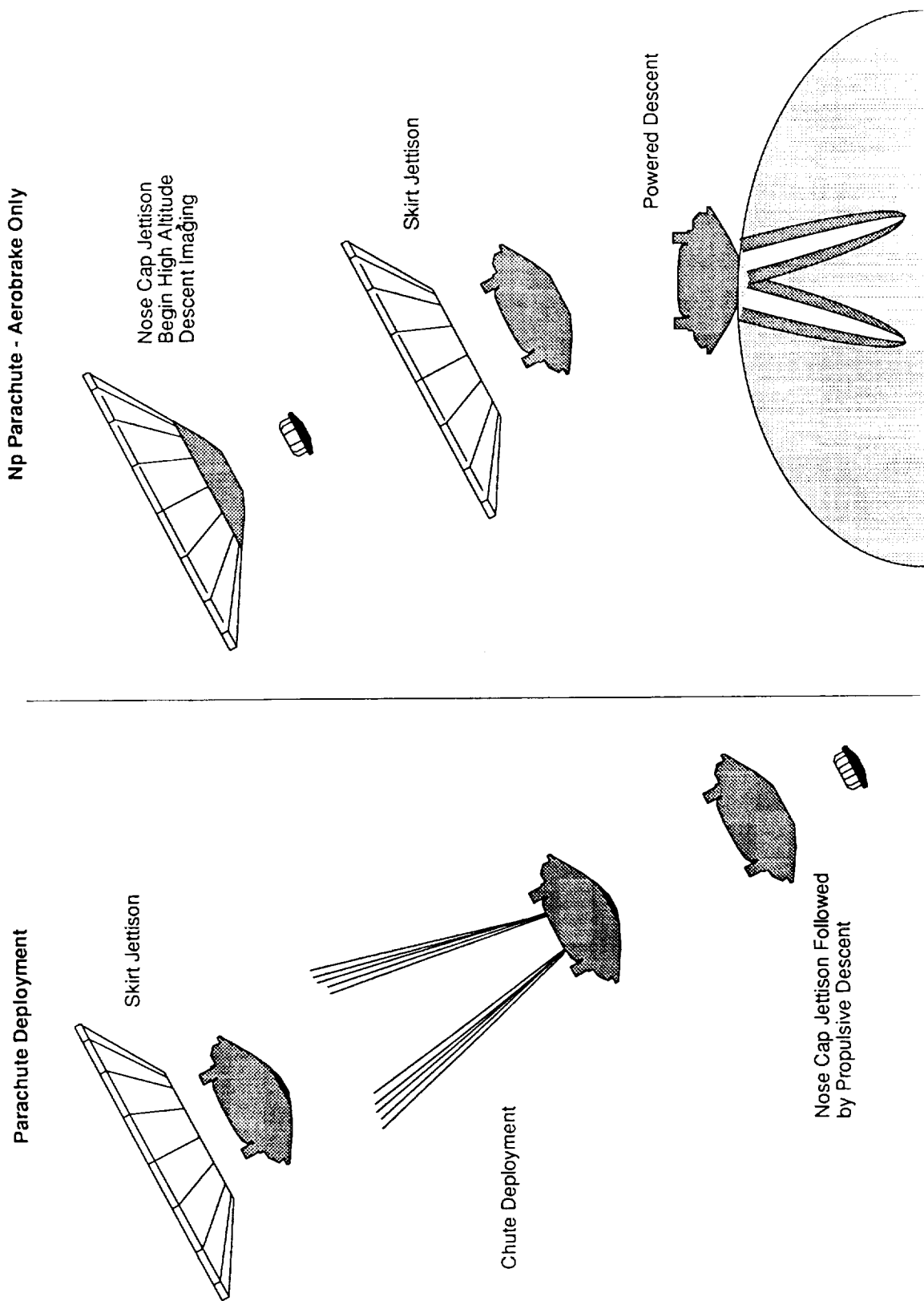
I ask the readers of this proposal who have become hardened by the decade long neglect of planetary exploration to try to suspend doubt in a sustained exploration program. Consider the GNM in a broader context of planetary exploration that has a new commitment behind it. If there is significant gold to be found in getting science value and the taxpayers money's worth in this program, it is in looking beyond the event horizon of the next mission.

I believe the GNM work shop was very productive and I look forward to future discussion of this and other promising mission and system design options for the Robotic Exploration Program.

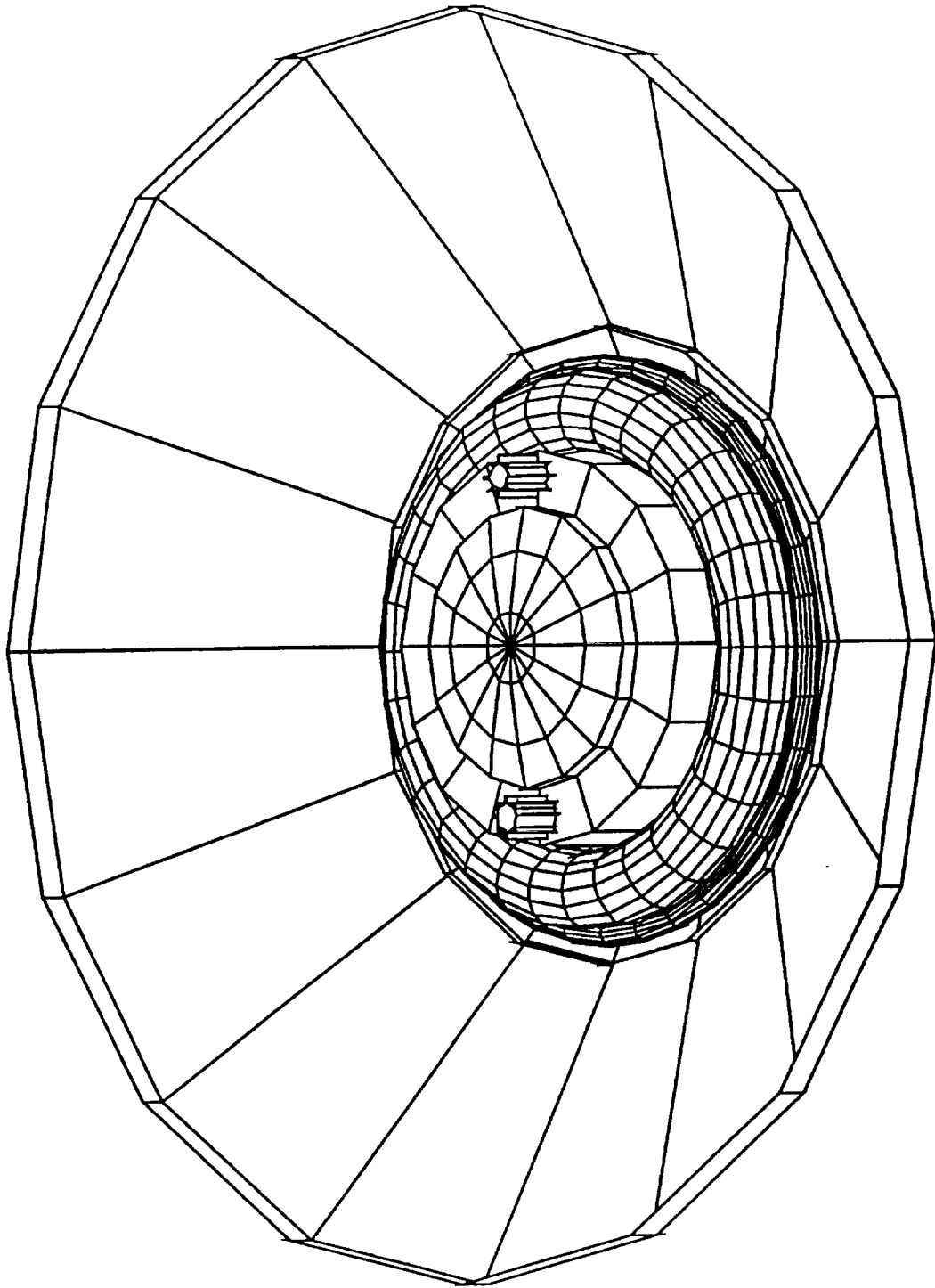
Proposed GNM Mission Scenario



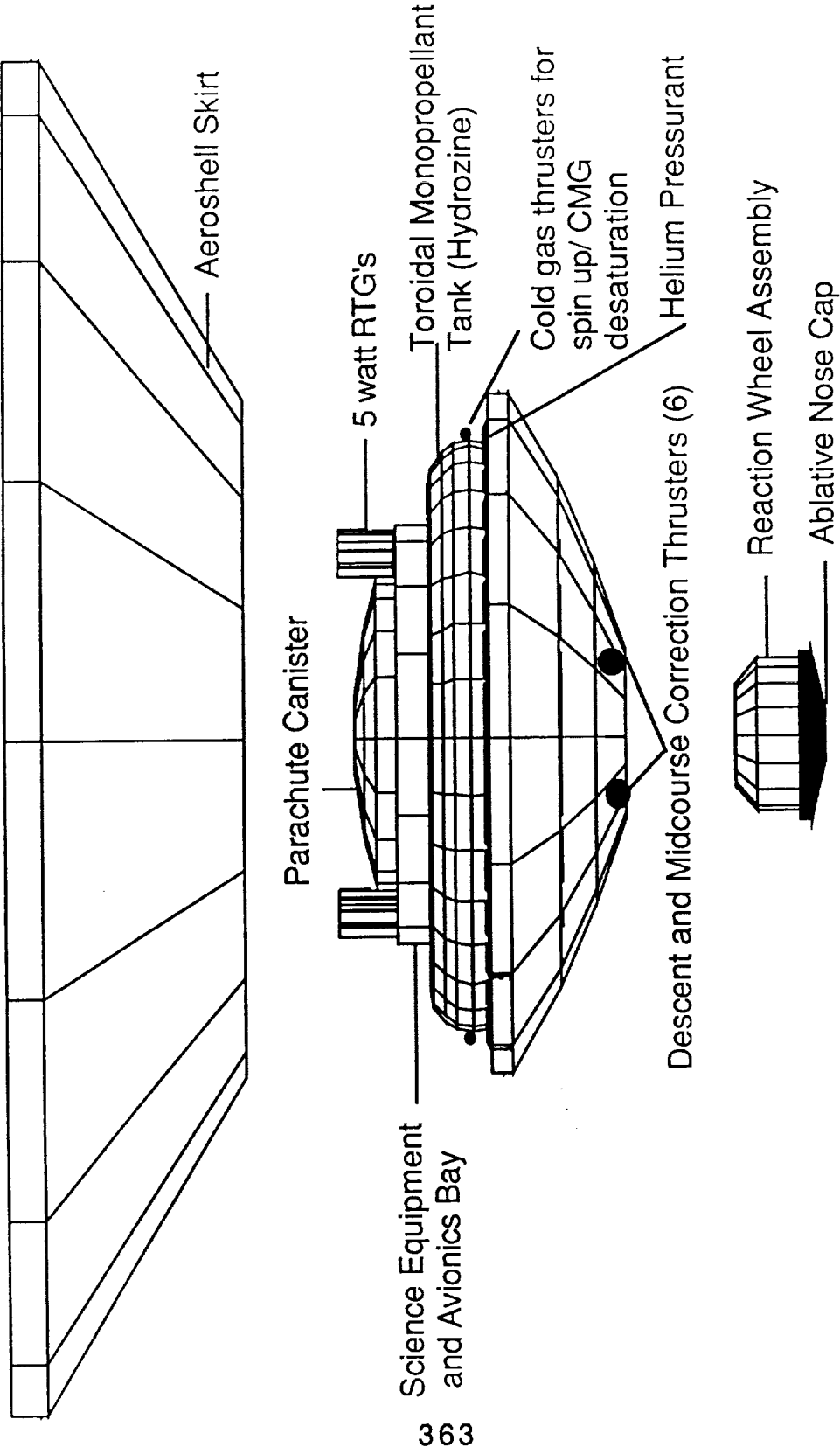
Terminal Descent Options



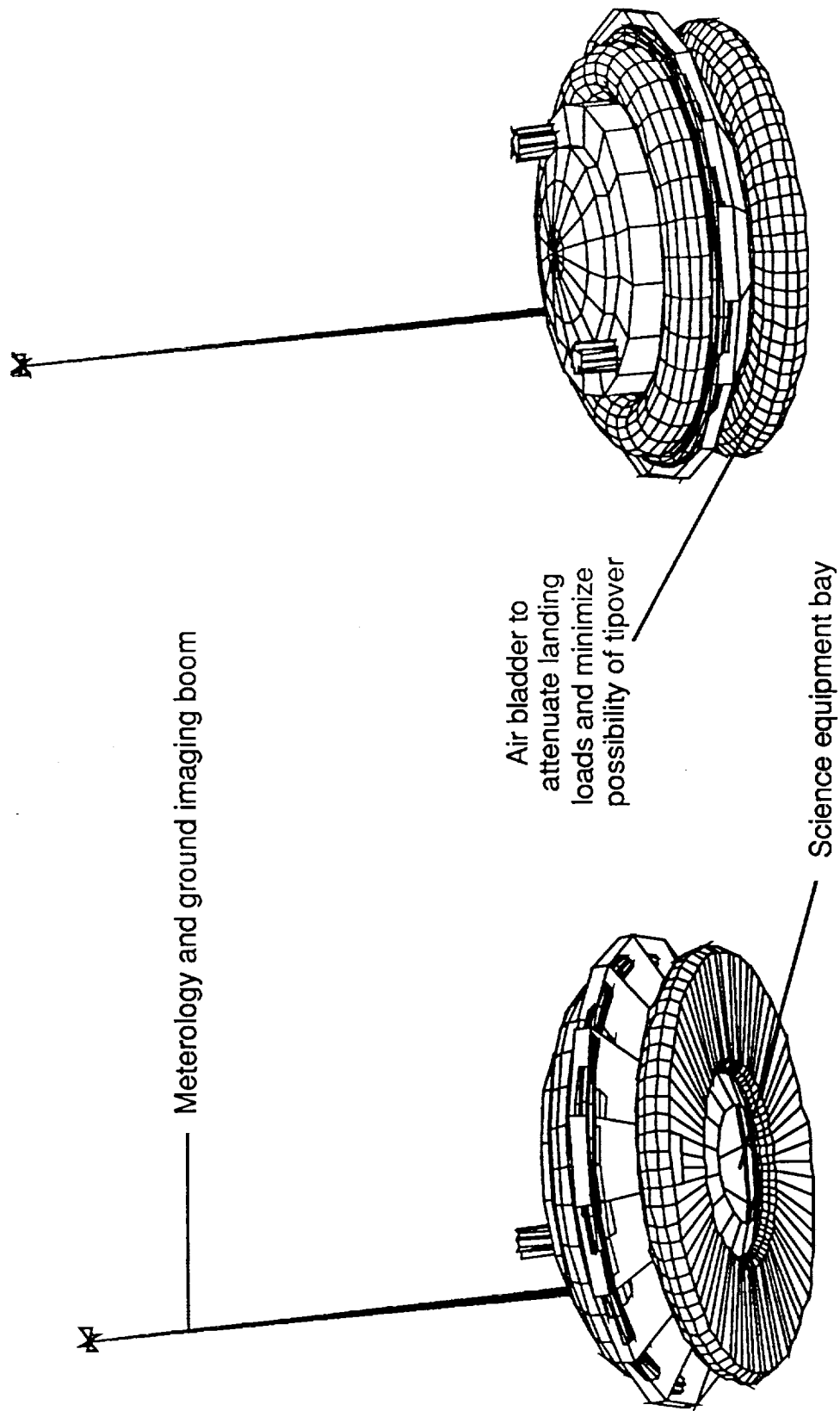
Aeroshelled Lander (Perspective)



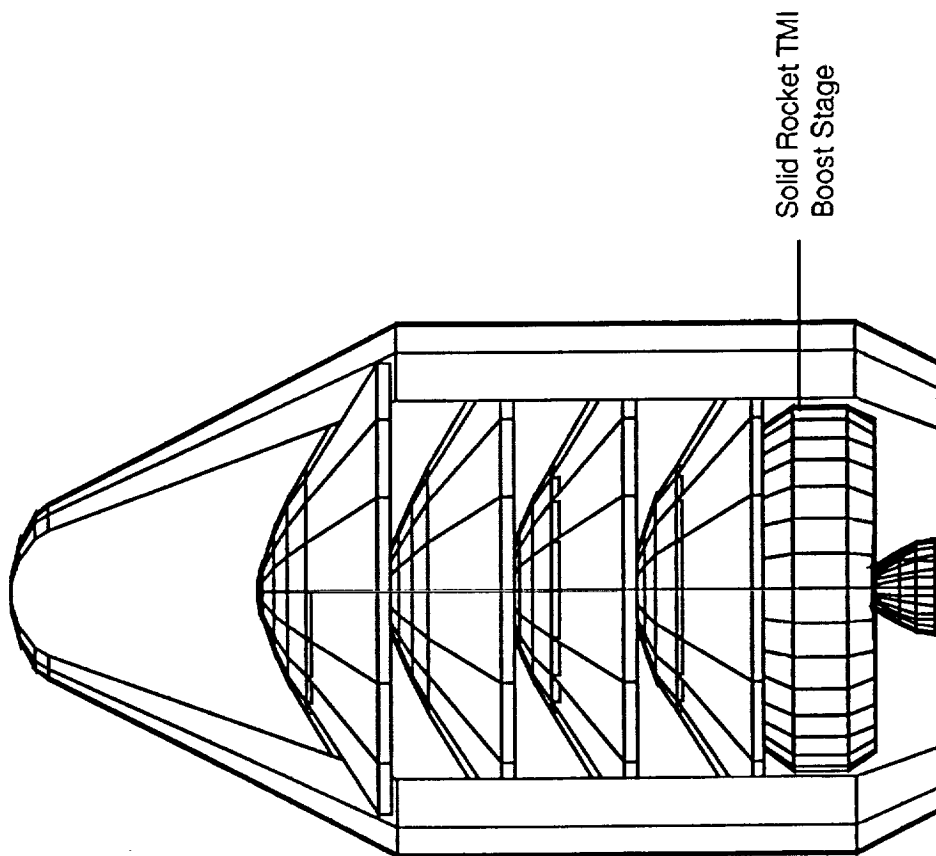
Simple Aeroshelled Lander Concept



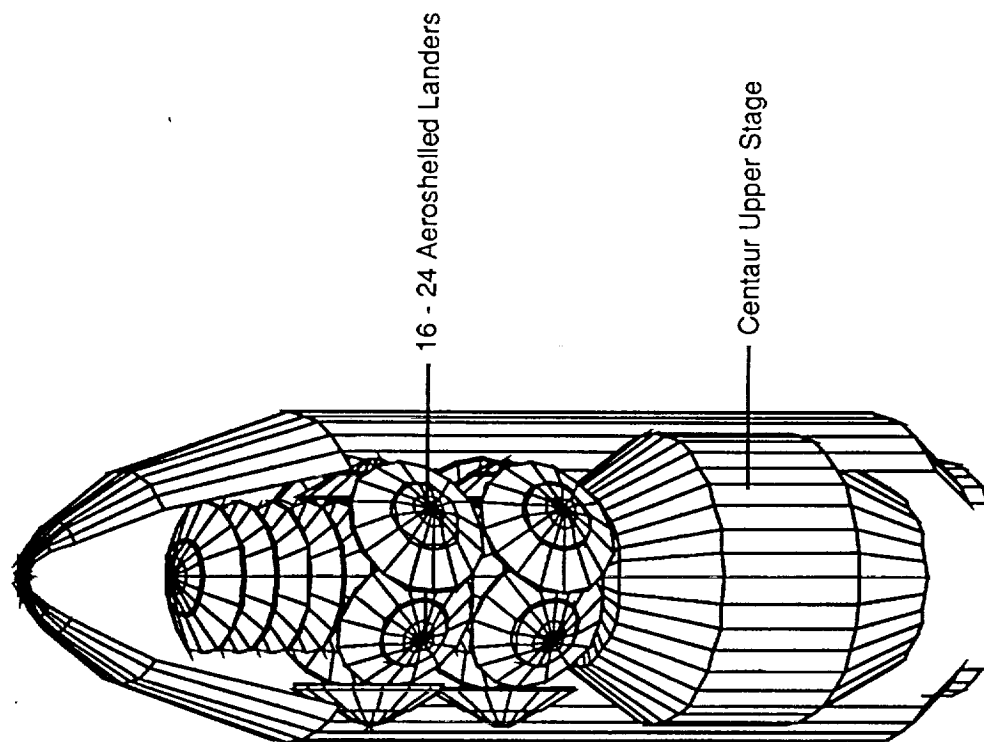
Lander Operational Configuration



Possible Delta Launch Configuration
3-4 Aeroshells per launch



Possible Titan IV Launch Configuration
15 - 24 Aeroshells per launch



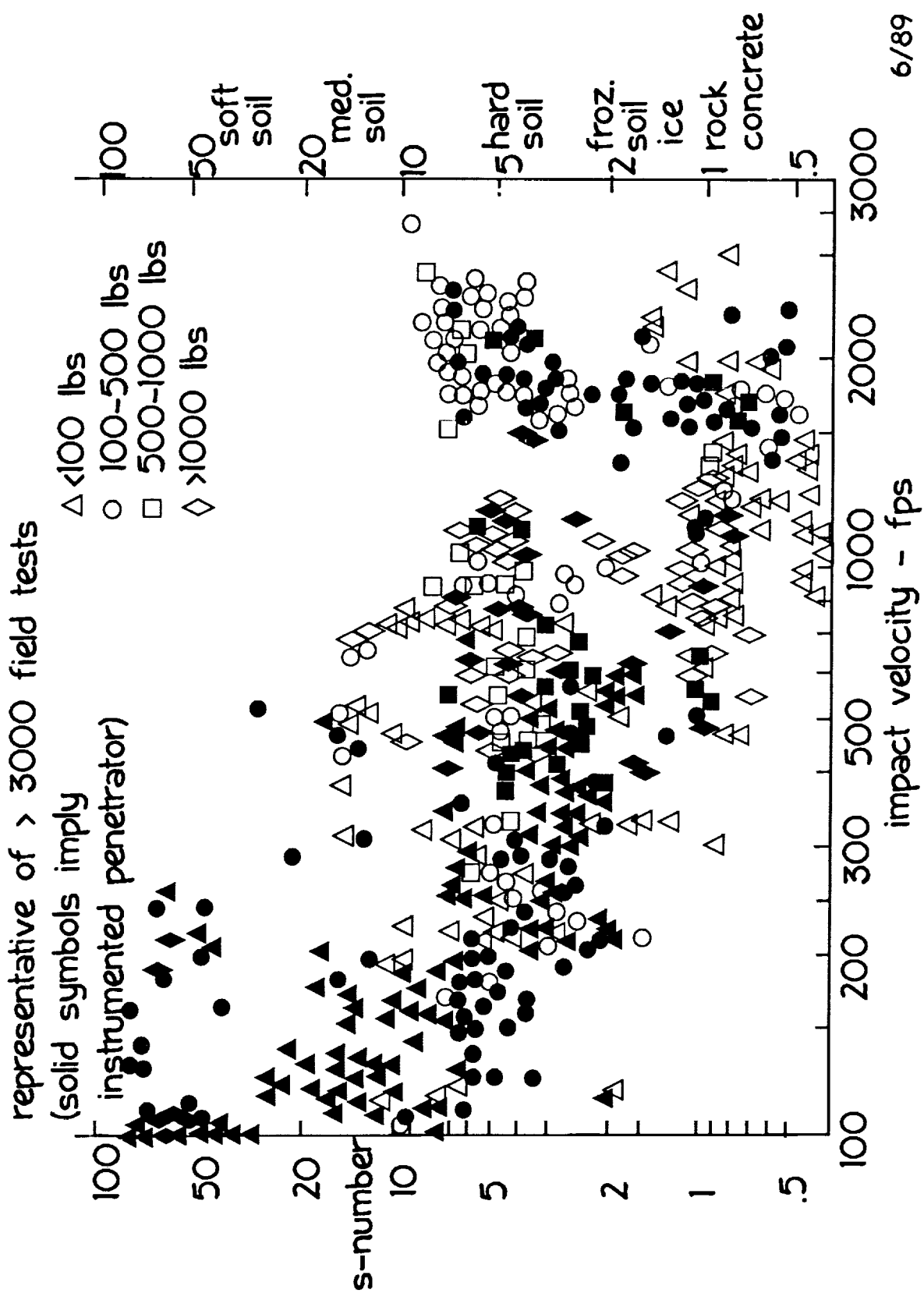
6.3 SESSION C SUBMITTALS

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Session C, Submittal No. 1

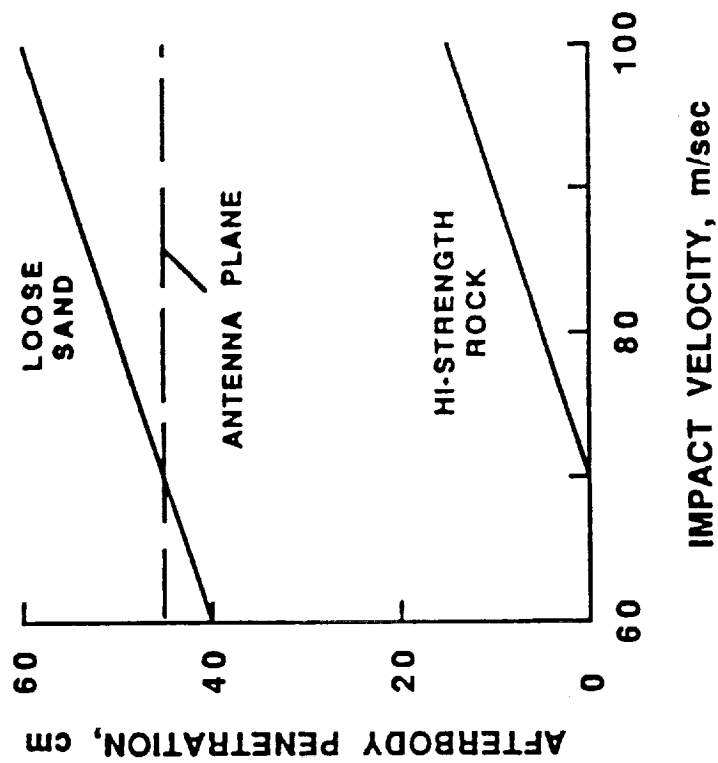
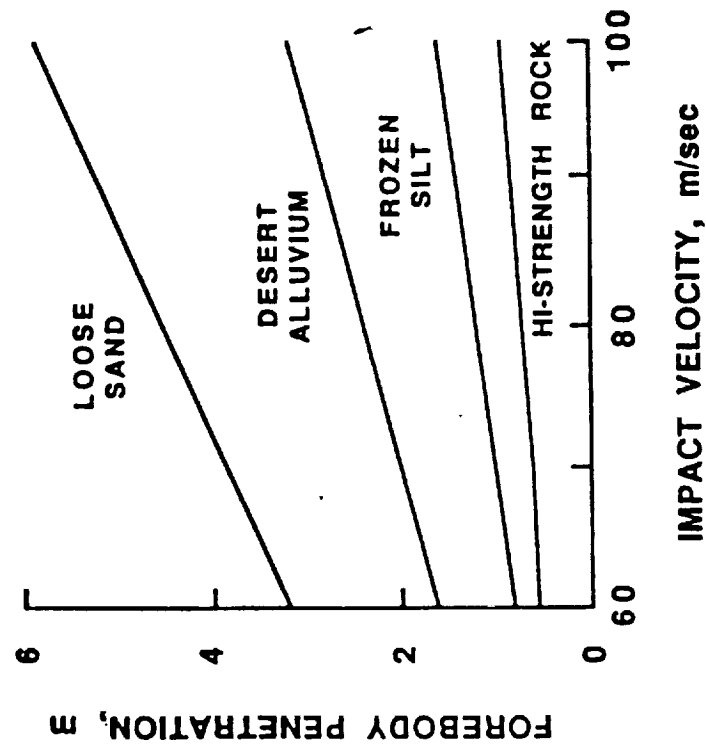
C. Wayne Young
Sandia National Laboratories

Sandia experimental earth penetration program

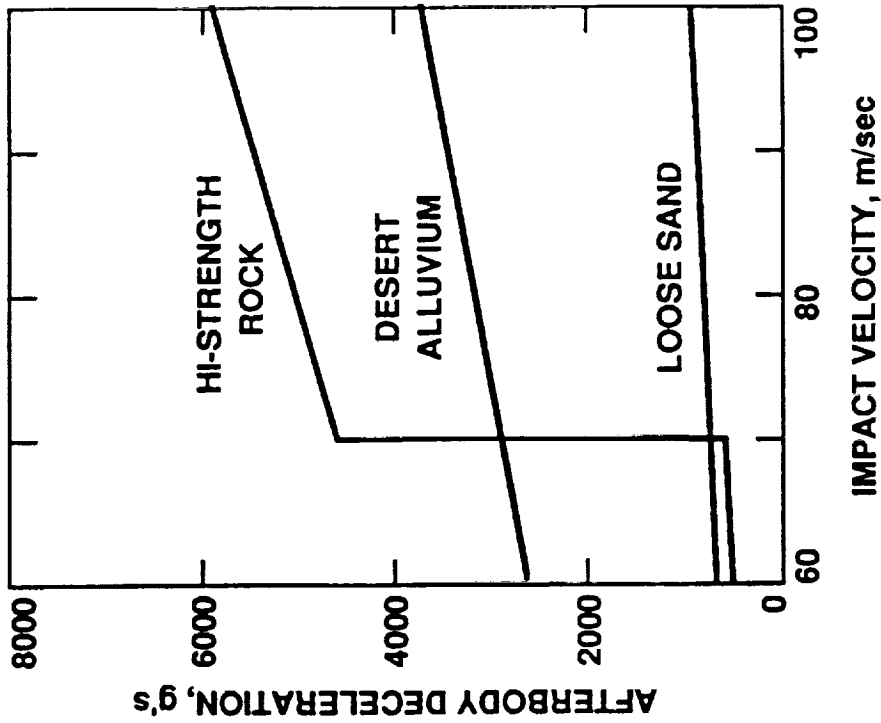
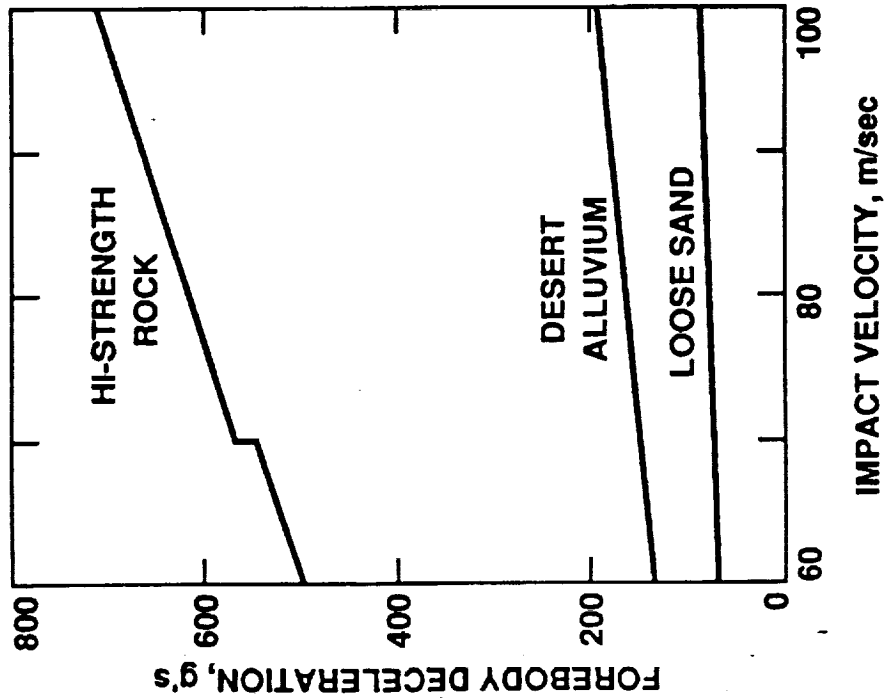


6/89

PENETRATION DEPTH



PENETRATION LOADS



Soil Penetrability

Unfrozen (use soil equation)

Very hard (dense and/or cemented sand, caliche, damp stiff silt or clay): $S = 5 \pm 2$

Medium hard (medium to loose sand, moist silt and clay): $S = 9 \pm 2$

Soft (topsoil, wet silt or clay): $S = 15 \pm 4$

Frozen (use ice equation)

Most moist to wet soils: $S = 2 \pm 0.25$

Dry soils: ?

Analytical Methods Recommended

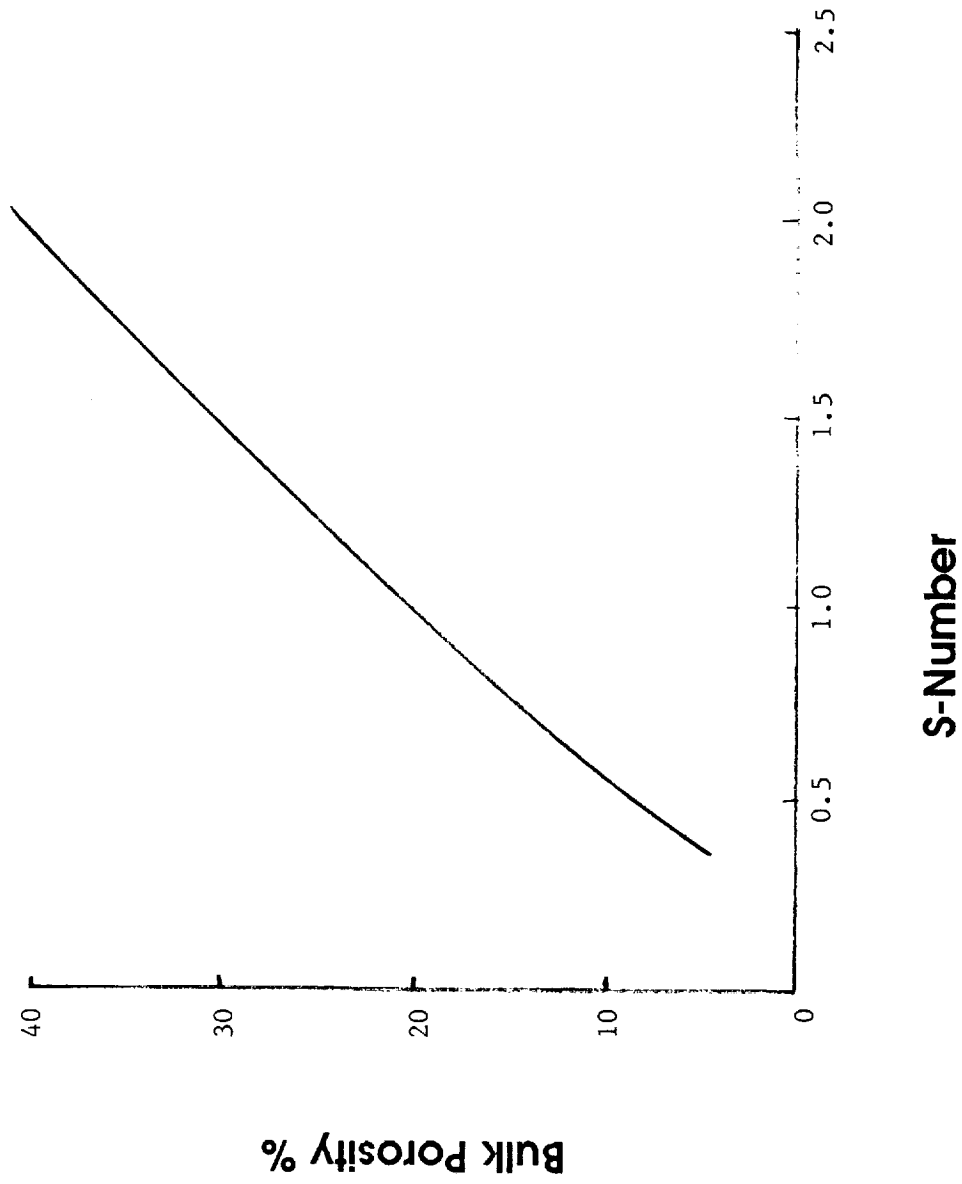
Axial Loading

- Sandia empirical equations
- Wavecodes are available, but perhaps not the most appropriate method

Lateral Plus Axial Loading

- SAMPLL - engineering tool
- Hydrocodes (PRONTO, DYNA, HULL, etc.) - more limited use, but necessary for coupled calculation with structural response

Rock Penetrability



Note: This is a useful guide, but not the only consideration.

Lateral Loading

NOTE: This will be critical element in penetrator design and component loading.

- Impact Angle
 - Angle of Attack
 - Target nonhomogeneity (borders, etc.)
 - SAMP LL, with approximations
 - Mostly experiments with instrumentation
- SAMP LL code & Hydrocodes**

Penetration Technology

Issues requiring early effort:

- **Target description in geologic terms**
- **Determine effect of boulders, etc.**
- **Evaluate effectiveness/utility of shock mitigation**
- **Suitability of Titanium for penetrator**

Conclusion: The technology exists to develop a penetrator with suitable structural and penetration performance. As with previous penetrating systems, this effort will require proper combination of experience, analysis, and experiment.

Session C, Submittal No. 2

David E. Ryerson
Sandia National Laboratories

**Sandia National Laboratories
Telemetry Department
High Shock Penetrator Instrumentation Program**

D. E. Ryerson
Division 5144
February 2, 1990

Sandia National Laboratories Telemetry Department has been building high shock instrumentation systems for penetration studies for over twenty years. The instrumentation systems are digital stored data acquisition systems used to gather data during the penetration event and then recovered for data readout. The systems are powered by batteries, which are presently Eagle Picher LTC-7PST thionyl chloride batteries.

The shock loads that these systems are designed for are:

20,000 g for 1 millisecond
8,000 g for 10 milliseconds
3,000 g for 20 milliseconds
1,000 g for 50 milliseconds

Sandia has been fielding an average of sixty instrumented penetrator tests per year for the last five years. Attached is a plot of a sample penetrator test acceleration record.

To make our electronics survive high shock, we constrain all of the components very tightly in the penetrator package. We use selected components and encapsulate them in hard potting per the attached "Rules for Building High-g Electronics." Our temperature environment is typically between 0 and 50 degrees Celsius, so we can use components that would not survive standard military temperature ranges.

We normally try not to use shock attenuation to protect the electronic components. An analysis of shock attenuation is given on an attached page. It shows that to get shock attenuation, one must let electronics move a much larger distance than the penetrator housing, which is impossible.

We have used material to remove high frequency components of a shock pulse to protect such devices as accelerometers which can be broken by high-amplitude high-frequency inputs. The disadvantage of this shock material is that it may distort the accelerometer response and in certain cases, actually amplify certain frequencies of the shock pulse. In our work, we stay away from shock attenuators if at all possible.

HIGH-G PENETRATOR INSTRUMENTATION

Sandia National Laboratories

Telemetry Department

1. Digital Stored Data Acquisition System
2. Shock Load Design Levels
 - 20,000 g for 1 millisecond
 - 8,000 g for 10 milliseconds
 - 3,000 g for 20 milliseconds
 - 1,000 g for 50 milliseconds
3. Components Tightly Constrained
4. Shock Attenuators Not Used

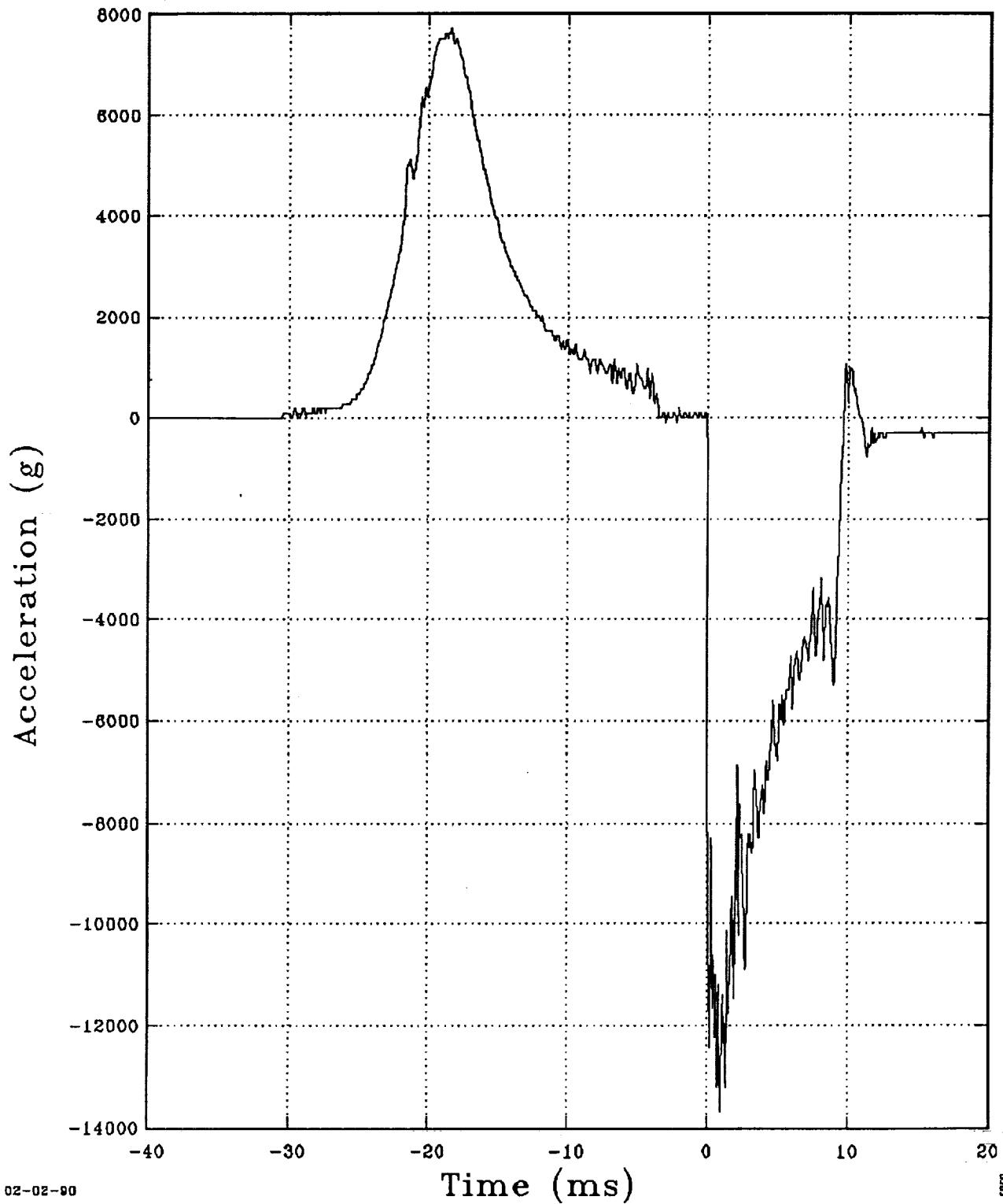
Sample Penetrator Test

Axial Acceleration

Analog LPF: 4800 Hz
Digital LPF: none

Test Date
07-13-88

SSP-86



02-02-80

SNLA
5144

RULES FOR BUILDING HIGH-G ELECTRONICS

**D. E. Ryerson
Sandia National Laboratories
Division 5144
February 2, 1990**

1. **Constrain the PC Boards and Components in Hard Potting** - Hard potting is required to keep components from moving during shock. Typical potting is epoxy filled with glass micro-balloons. Make sure electronics and potting material are compatible with temperature ranges that the system will see in curing of potting and system operation.
2. **Cover the Components with a Thin Layer of Soft Potting** - Soft potting protects components during the hard potting curing process. It also gives a slight cushion to the component. Typical potting used is polysulfide rubber. Some silicone-type materials will not work because they act like mold release and will not let the hard potting adhere to the boards.
3. **Use as Small a PC Board as Possible** - The smaller a board is the less likely it is going to flex and break.
4. **Mount Small Components such as Resistors and Diodes Away from PC Board** - Small components can be broken by a board which flexes, especially if the board has raised solder mounds or lands under the component.
5. **Mount Shims Between Integrated Circuits and PC Boards** - Potting will typically not flow under an IC and a void will be left. Voids or air pockets allow components to move and break.
6. **Interconnect PC Boards with Fixed Wires or Spring Sockets and Beryllium Wire** - Normal connectors are prone to break.
7. **Use Plastic Integrated Circuits** - Plastic integrated circuits have the wires running from the IC pins to the die encapsulated. Ceramic IC's leave a cavity for the wires and die. The small wires will often move and short out during shock in a ceramic IC.
8. **Do Not Use Large Electrolytic Capacitors - Use Only Ceramic Capacitors** - Many large electrolytic capacitors cannot take shock. Solid electrolytic capacitors such as Kemet parts may work. Avoid large capacitors if possible. If that cannot be done, test components under shock to determine survivability.
9. **Use Small Known High-Shock Batteries** - Large batteries typically have internal construction which will not survive shock. Test battery types under shock to determine survivability.
10. **Do Not Overcharge Batteries** - When batteries are overcharged or charged too fast, they will expand and crack the case or potting which holds the cells.
11. **Keep Power Consumption Low** - Keep the system power consumption low to keep battery size down.
12. **Preload Package when Mounting in Hardware**
13. **Present Major Shock Perpendicular to PC Board Instead of Along Board**

RULES FOR BUILDING HIGH-G ELECTRONICS

Sandia National Laboratories

Telemetry Department

1. Constrain PC Boards and Components in Hard Potting
2. Cover Components with a Thin Layer of Soft Potting
3. Use Small PC Boards
4. Mount Discrete Components Away from PC Board
5. Mount Shims between IC and PC Board
6. Minimize Connector Use
7. Use Plastic ICs
8. Don't Use Large Electrolytic Capacitors
9. Use Small Batteries
10. Don't Overcharge Batteries
11. Keep Power Consumption Low
12. Preload Package in Mounting Hardware
13. Shock PC Boards Perpendicularly

Shock Attenuation

D. E. Ryerson
Sandia National Laboratories
Division 5144
February 2, 1990

The purpose of a shock attenuator is to reduce the amplitude of a deceleration pulse. Assume a deceleration pulse of constant amplitude A for time T . Calculate the motion parameters as follows:

$$\text{acceleration} \quad - a = A \text{ for time } T$$

$$\begin{aligned} \text{velocity} \quad - v &= -V_0 + \int a \, dt = -V_0 + A t, \quad 0 < t < T \\ V_0 &= A T \text{ to force } v = 0 \text{ at } t = T \\ v &= A (t - T), \quad 0 < t < T \end{aligned}$$

$$\begin{aligned} \text{depth} \quad - d &= -\int v \, dt = -A (1/2 t^2 - T t), \quad 0 < t < T \\ d &= 1/2 A T^2, \quad t = T \end{aligned}$$

A shock attenuator would reduce the deceleration by slowing the body over a longer time interval. Let's calculate the energy in the shock pulse and hold it constant as follows:

$$\text{energy} = E = \text{force} * \text{distance} = \text{mass} * \text{acceleration} * \text{distance}$$

$$E = m A 1/2 A T^2 = 1/2 m (A T)^2$$

$$\text{let } E_2 = E_1 \quad \rightarrow$$

$$A_2 T_2 = A_1 T_1$$

$$\text{since } d = 1/2 A T^2 \text{ and } (A_2 T_2)^2 = (A_1 T_1)^2$$

$$\rightarrow$$

$$A_2 d_2 = A_1 d_1$$

Therefore, the time of the deceleration pulse is inversely proportional to the amplitude of the pulse to keep the energy in the pulse constant and the depth of penetration is also inversely proportional to the deceleration amplitude.

Summary

A shock attenuator must allow the device being decelerated to travel over a longer distance to get any shock attenuation. If the device is being stopped in centimeters, it may be possible to double the stop distance to halve the deceleration. If the device is being stopped in meters, it probably is not possible to double this stop distance.

In penetrator work at Sandia, we have found that shock attenuators do not work to protect our electronics. We have found that in some cases an elastic medium has been useful in removing the high frequency components or fast rise times of the deceleration pulse. If one is not careful, it is possible that such elastic media will become shock amplifiers at certain frequencies (resonances) rather than shock attenuators.

DECELERATION SHOCK ATTENUATION

Sandia National Laboratories

Telemetry Department

1. Assume Constant Deceleration of Amplitude (A)
for Time (T) seconds
for Depth (D) meters
2. For Constant Energy in Shock Pulse
Amplitude (A) * Time (T) = constant
Amplitude (A) * Depth (D) = constant
3. To Significantly Reduce Shock Amplitude,
Depth Must Be Significantly Increased
4. Shock Attenuators Have Not Proven Feasible
in Sandia's Penetrator Program

David E. Ryerson

Supervisor of Telemetry Technology Development Division 5144 at Sandia National Laboratories, Albuquerque, New Mexico.

BS in Electrical Engineering, Iowa State University, 1965.

MS in Electrical Engineering, University of New Mexico, 1967.

Worked at Sandia from 1965 to the present in telemetry, data acquisition, and control systems. Designed real-time aircraft computer-controlled systems for target tracking and rocket-launch computer systems for Sandia's Kauai test range. Developed long-life (1 to 3 years) ocean-floor seismic systems and underwater acoustic telemetry for data recovery. Presently directing the designing and fielding of ultra-high shock (up to 20,000 times gravity) penetrator data acquisitions systems, rocket and reentry vehicle instrumentation, and specialized data acquisition systems.

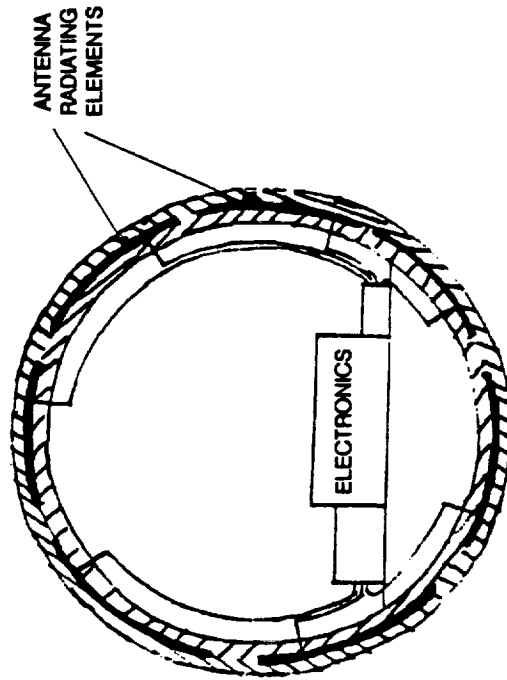
Session C, Submittal No. 3

Tomas A. Komarek
Jet Propulsion Laboratory/California Institute of Technology

PENETRATOR RF HARDWARE CONCERNS

- **PACKAGING METHODOLOGY FOR SURVIVAL OF SHORT-DURATION, HIGH - IMPACT DECELERATIONS (TO 10,000 *g*'s).**
- **THERMAL DESIGN OF RF/ELECTRONIC CIRCUITS FOR A WIDE RANGE OF SURFACE TEMPERATURES.**
- **LONG LIFE TIME (3 – 5 YEARS) FOR MANY DEEP TEMPERATURE CYCLES AFTER HIGH-IMPACT SHOCK.**
- **RELIABLE RF CIRCUIT PERFORMANCE WITH STABILITY FOR THE ANTICIPATED DIVERSE ENVIRONMENTS.**
- **CRYSTAL OSCILLATORS (RECEIVER, EXCITER, CDU, TMU).**
 - **CRYSTAL FILTERS FOR NARROW - BAND RECEIVERS.**
 - **FILTERS/DIPLXERS (RF SYSTEM).**
 - **RF SWITCHES FOR REDUNDANCY SWITCHING.**
 - **RECEIVERS, EXCITERS, TRANSMITTERS, ANTENNAS AND RF COMPONENTS.**
- **VOLTAGE?**
- **TRANSMITTERS.**
- **CONCLUSION: ATD NEEDED TO EVALUATE AND IMPROVE DESIGNS.**

HIGH-IMPACT SPHERICAL SHELL ARRAY ANTENNA



- SURVIVE UP TO 10,000 *g* IMPACT.
- BEAM SWITCHING OR BEAM SCANNING
- GRAVITY-ASSISTED DIRECTION FINDING.
- UHF.
- APPROXIMATELY 10-dB GAIN.
- APPROXIMATELY 14 RADIATING ELEMENTS.
- APPROXIMATELY 12-IN. IN DIAMETER.

Session C, Submittal No. 4

Farley Palmer
Hughes Aircraft Company

Session C - Subsystem Technology Focal Questions



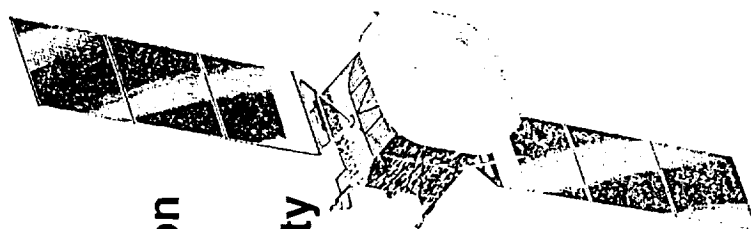
- 1) What technology will help achieve 10 year life-times?
- 2) What technology will help survival of high-g landings?
- 3) Are RTGs a workable power subsystem (size, location on the lander)?

Achieving 10 Year Lifetimes

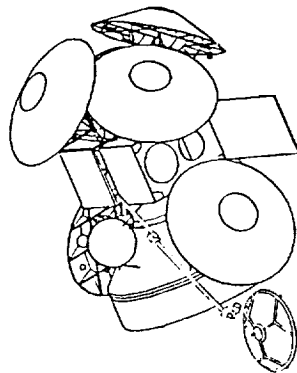
HUGHES
AIRCRAFT COMPANY

Terrestrial communications
satellites designed for >10
years today

- o Life = 14 yrs
- o Mean Mission Duration = 12.6 yrs at 0.74 reliability



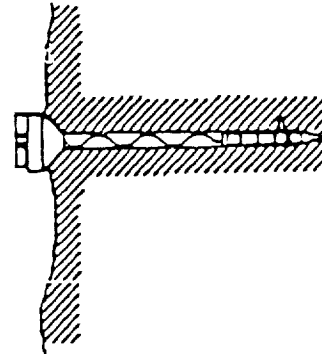
Direct application of contemporary
satellite design practice



CARRIER

- o Identify requirements
- o Design
- o Materials/parts
- o Reliability analysis
- o Redundancy
- o Qualification test

Cross-application of contemporary satellite design practice



PROBE

ISSUES

- o Requirements?
- o Which objectives require long life?
- o Can they be separated?
- o High-g survivability
- o Unique Martian environmental effects

UHF FOLLOW-ON

Mars Global Network Mission Science & HEI Functions

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AIRCRAFT COMPANY

Science or Function	Location	g-Hard Requirement	Life Requirement
Chemistry Entry Hydrology Imagery Impact Meteorology Mineralogy Navigation beacon Seismometry Sub-surface experiments Surface experiments	Top-side/Sub Inbound Sub-surface Inbound/Top Sub-surface Inbound/Top Top-side/Sub Top-side Sub-surface Sub-surface Top-side	Possible No Yes No Yes No Yes & No No Yes Yes No	Short Ultra short Short Short->Medium Ultra short Medium->Long Short Very long Long Short Short->Long

Probe Issues

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AIRCRAFT COMPANY

Science goals

Human Exploration Initiative (HEI) support & infrastructure goals

System optimization

Single-design vs multi-design

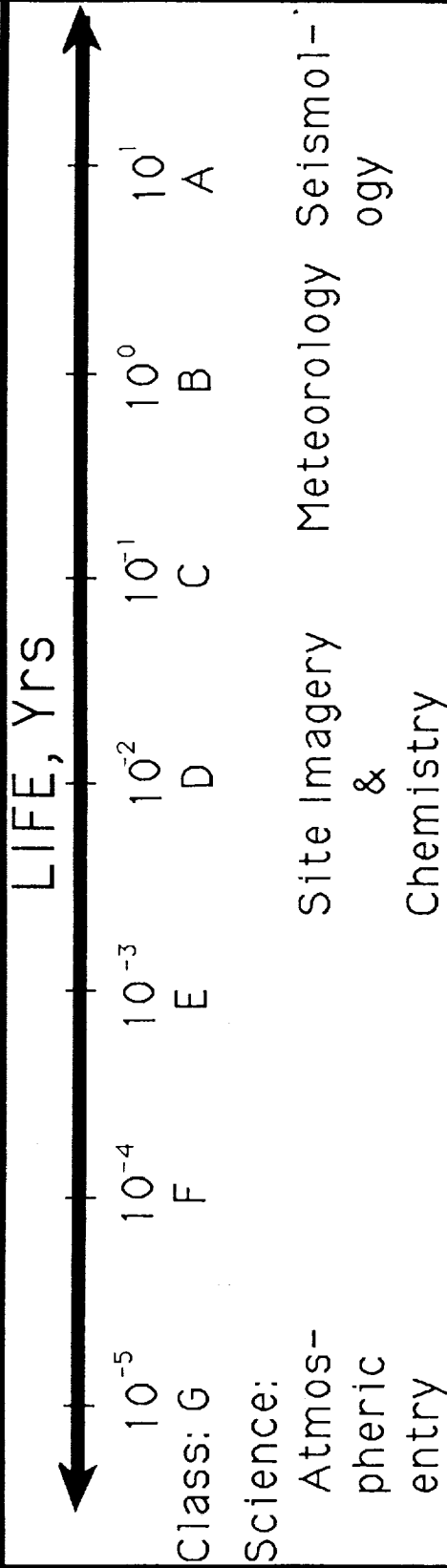
- Penetrator vs Penetrator & surface lander
- Long life vs Long & short life probes
- Large mass vs medium & low mass probes

Surface lander impacts vs soft-lands

Aeroshell vs propulsive deceleration

Trades Associated with Life Power System Example

HUGHES
AIRCRAFT COMPANY



- Do all science objectives require integration in a single probe?
- What power availability is required?
 - Guaranteed vs Intermittant-but-statistically-probable
- Is the requirement "baseline," "bi-modal," or "transient periodic/aperiodic?"

Candidate Power System Approaches



Class	Type	Availability	Burst Power	Example
A1	Bi-modal	Guaranteed	Oversized primary with or without secondary storage	<ul style="list-style-type: none"> Primary battery RTG + 2ary battery or flywheel
A2	Bi-modal	Intermittent but statistically probable	Oversized primary with or without secondary storage	<ul style="list-style-type: none"> Solar panel or wind generator + 2ary battery or flywheel
B1	Baseload	Guaranteed	N/A	<ul style="list-style-type: none"> Primary battery RTG
B2	Baseload	Intermittent but statistically probable	N/A	<ul style="list-style-type: none"> Solar panel Wind generator
C1	Transient	Guaranteed	Possible secondary storage	<ul style="list-style-type: none"> Primary battery RTG + 2ary battery or flywheel
C2	Transient	Intermittent but statistically probable	Possible secondary storage	<ul style="list-style-type: none"> Solar panel or wind generator + 2ary battery or flywheel

High-g Survival



SDI Electromagnetically Launched Guided Projectiles

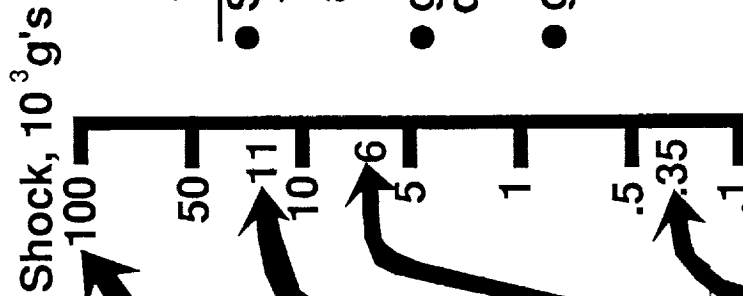
- (LEAP)

Artillery-Tube-Launched Guided Projectiles

- 155 mm Laser homing (Copper-head)
- 81 mm Infrared homing mortar bomb

Celestial Body Probes & Impactors

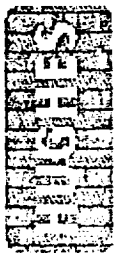
- Mars Global Network Mission
- European asteroid or comet penetrator
- Galileo Probe to Jupiter



High-g Technology

- Super-scale integration to reduce mass and size
- g-hard subsystem designs
- g-hard packaging

Summary and Conclusions



- Achieving 10-year-life spacecraft is feasible
- Life concerns believed exclusive to probe
 - Power system (eg: RTG)
- Precedent exists for g-hard probe design
 - "Brilliant pebbles"-like guided missiles
 - Requires strong system integrator role

Session C, Submittal No. 5

Michael Shirbacheh
Jet Propulsion Laboratory/California Institute of Technology

(Information provided by Wayne M. Brittain,
Teledyne Energy Systems)

**SPECIAL APPLICATIONS LOW-POWER RTG
DEVELOPMENT PROGRAM SUMMARY**

January 1990

SPECIAL APPLICATIONS LOW-POWER RTG

DEVELOPMENT PROGRAM SUMMARY

A. Background

The Special Applications RTG Development Program was initiated at Teledyne Energy Systems (TES) in September 1983 under DOE Contract DE-AC01-83NE32115. The development effort was performed under this contract through September 1988. After this time the program was continued as the Two-Watt Special Applications RTG Program (DOE Contract DE-AC01-88NE32142) with the objective of fueling a prototype RTG unit. Present activities at TES include fabrication, assembly and test of the electrically-heated prototype RTG which will be delivered to EG&G/Mound in June 1990 for fueling in December 1990.

Development of a sealed, 3-layer fuel capsule for use in the Two-Watt RTG is being performed for DOE in a joint effort by TES, EG&G/Mound and LANL. The capsule design is based on an upsizing of the Milliwatt RTG and Navy One-Half Watt RTG terrestrial 3-layer capsule technology.

B. Introduction

The primary objectives of the Special Applications RTG Development Program are to:

- (1) develop a low-power (2 to 5W) relatively high voltage (5 to 12V) thermoelectric module using proven PbTe/TAGS thermoelectric materials. This materials technology has been applied to both NASA SNAP-19 space RTGs (Pioneer 10 and 11 Jupiter Fly-by spacecraft and Viking 1 and 2 Mars Landers), and terrestrial RTGs delivered to DOE

- for subsea applications. Demonstrated thermoelectric module technology for low-power terrestrial RTGs at the initiation of the development program was limited to bismuth telluride with a typical RTG system efficiency of 3.5 to 4.0%. The goal for the development program was to increase this efficiency by 50%.
- (2) develop a sealed heat source intended for terrestrial applications to contain the helium gas generated by the Pu-238 fuel decay. Available RTG heat source technologies for the anticipated thermal inventory requirement were all vented designs which result in increased parasitic heat losses with operating time due to the introduction of helium into the thermal insulation. The goal was to contain this helium within the capsule.
 - (3) design, fabricate, assemble, fuel and test a prototype terrestrial RTG system to demonstrate the developed technology. The selected terrestrial RTG design would consider potential near-term applications of low-power RTGs.

Although the hardware development for the Special Applications RTG has been oriented towards terrestrial applications, the thermoelectric module technology is generic and may be adapted to both space and terrestrial missions which require a low-power RTG power source. The radioisotopic heat source for space applications can be selected from available, qualified space hardware (such as the GPHS technology) or possibly be specifically designed and qualified for the mission requirements.

C. Thermoelectric Module Technology Description

The Special Applications thermoelectric module has evolved during the development program from using a couple with an all-PbTe N-leg and TAGS P-leg to one with Bi_2Te_3 cold segments on both the N and P-legs. The Bi_2Te_3 cold segments were added to the latest generation of thermoelectric modules to enhance the thermoelectric conversion efficiency for terrestrial applications where the RTG would be exposed directly to the cold subsea environment. These cold segments would not be beneficial for space applications and would not be included in the thermoelectric couple design.

1. Viewgraph 1

Viewgraph 1 shows Special Applications PbTe/TAGS minicouples which exemplify a configuration which could be considered for space application. The couple design is basically a miniaturization of the proven SNAP-19 space RTG thermoelectric technology. The couple has iron hot and cold shoes and copper pins to provide for electrically interconnecting the couples within a module. The Special Applications module uses a printed circuit board at the cold side to complete the interconnects between the couples. For the couple shown the individual legs are 0.102 in. sq. by 0.625 lg.

2. Viewgraph 2

Viewgraph 2 shows the typical internal construction of a Special Applications RTG. The configuration shown is that for the subsea prototype RTG

now being fabricated at TES. The 30-pound weight shown is almost all in the BeCu pressure housing, with less than 5 pounds attributable to the RTG internal components (thermoelectric module, heat source, heat distribution cup, thermal insulation and preload springs). For a space RTG configuration, particularly for a penetrator mission with high shock loading, the internal configuration would probably vary somewhat from that shown to satisfy mission vibration/shock requirements. For example, the heat source could have a support system independent of the thermoelectric module to minimize dynamic loads on the module.

3. Viewgraph 3

Viewgraph 3 shows a typical Special Applications thermoelectric module containing 68 couples. The module is approximately 3 inches in diameter by 0.8 inch thick. The cold side printed circuit board provides the basic structure for the module. Powdered Min-K thermal insulation is vacuum-impregnated between the couples to minimize heat loss. A thermoelectric module similar to that shown has been successfully tested to a 100g axial, 50g lateral (both applied simultaneously) shock loading to simulate impact deployment of an RTG.

4. Viewgraph 4

Viewgraph 4 shows the typical performance for the subsea RTG design shown on Viewgraph 2. The BOL in-water power output is approximately 5W with a system efficiency approaching 7%. Note that the hot junction temperature of the

terrestrial RTG is limited by the 3-layer capsule technology which has a long-term operating limit of approximately 1100°F. For a space application the hot junction would probably be increased to the 950°F range to take advantage of the high temperature heat source.

5. Viewgraphs 5, 6 & 7

Viewgraphs 5, 6 and 7 depict an alternate module configuration developed on the Special Applications program called a "Close-Packed-Array" or CPA. These viewgraphs show the configuration and performance of a 30-couple module rated at approximately 1.2W power output at 2.4V load voltage.

6. Viewgraphs 8, 9 and 10

Viewgraphs 8, 9 and 10 depict a module with a construction similar to that of the 30-couple module previously shown rated at 4.2W power output at approximately 6V load voltage.

7. Viewgraphs 11, 12 and 13

Viewgraphs 11, 12 and 13 show a 5W level module at approximately 9V load voltage. The module has 126 couples.

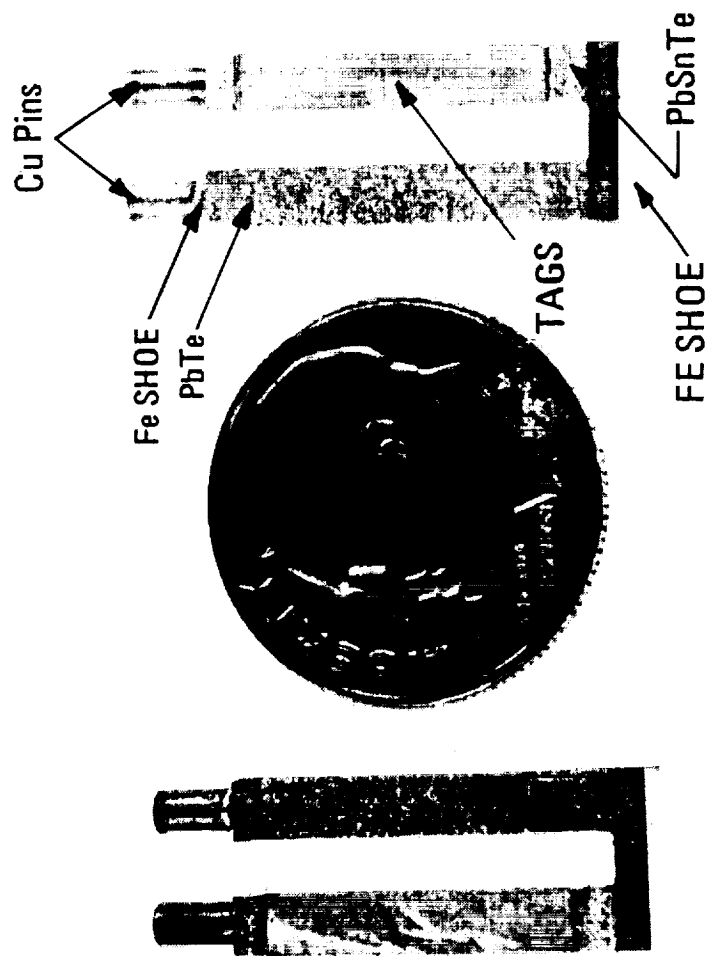
8. Viewgraph 14

Viewgraph 14 shows the conceptual design for a 10-15 W (at 9-12V) space RTG generated for a potential DOD space application using minicouples in conjunction with a 250W thermal GPHS heat source module. This concept uses the conventional SNAP-19 spring/piston cold end hardware arrangement to individually spring-load

each leg of the thermoelectric couple. This arrangement is an alternate to that shown in Viewgraph 2 where the thermoelectric module is loaded as a unit with preload springs.

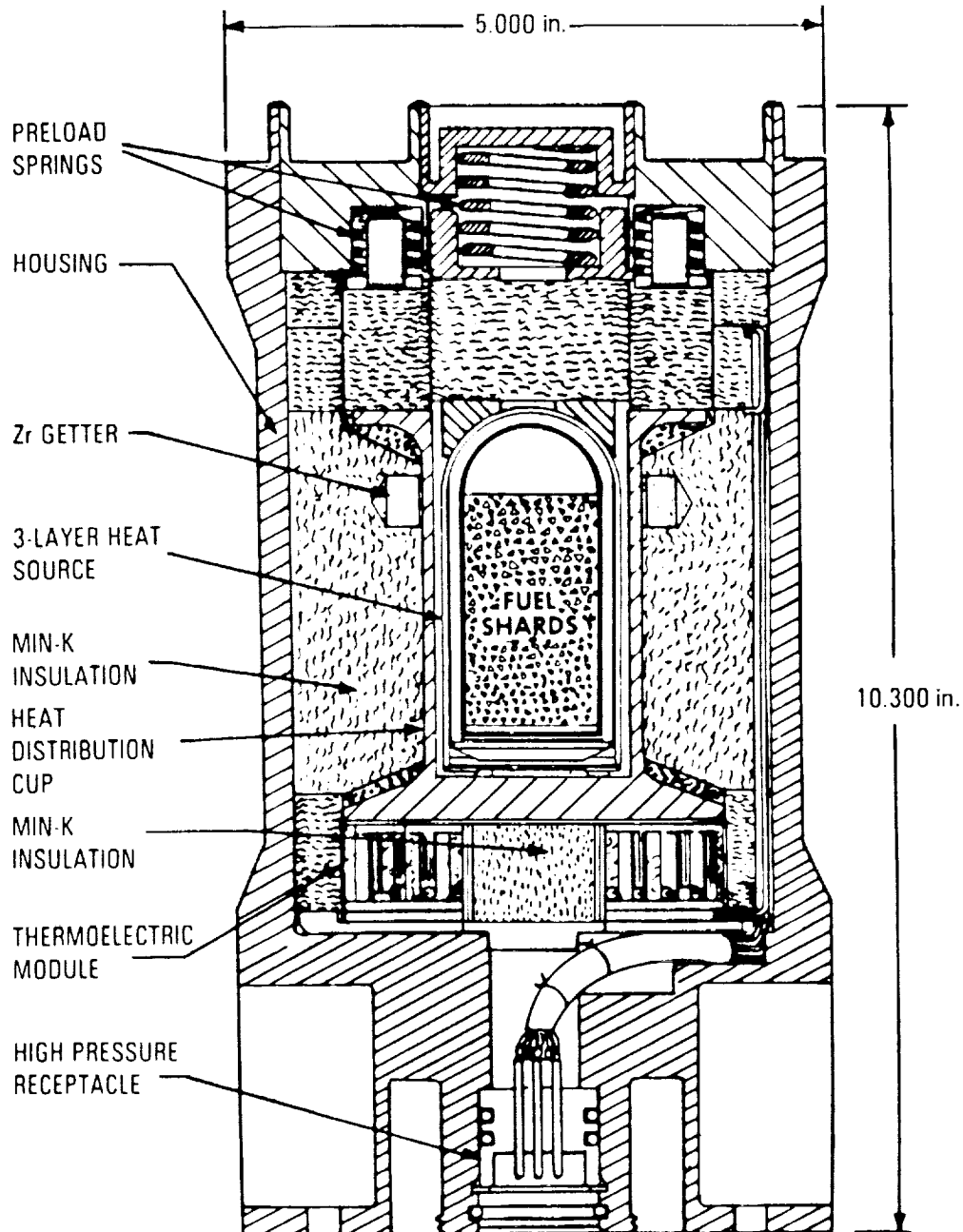
In summary, the Special Applications thermoelectric module technology is flexible both in its configuration and power level, permitting its adaptation to both space and terrestrial RTG missions requiring low-power RTGs. The RTG configuration and internal component support structure design would depend on the specific mission requirements.

SPECIAL APPLICATIONS THERMOELECTRIC MINICOUPLE



Viewgraph 1

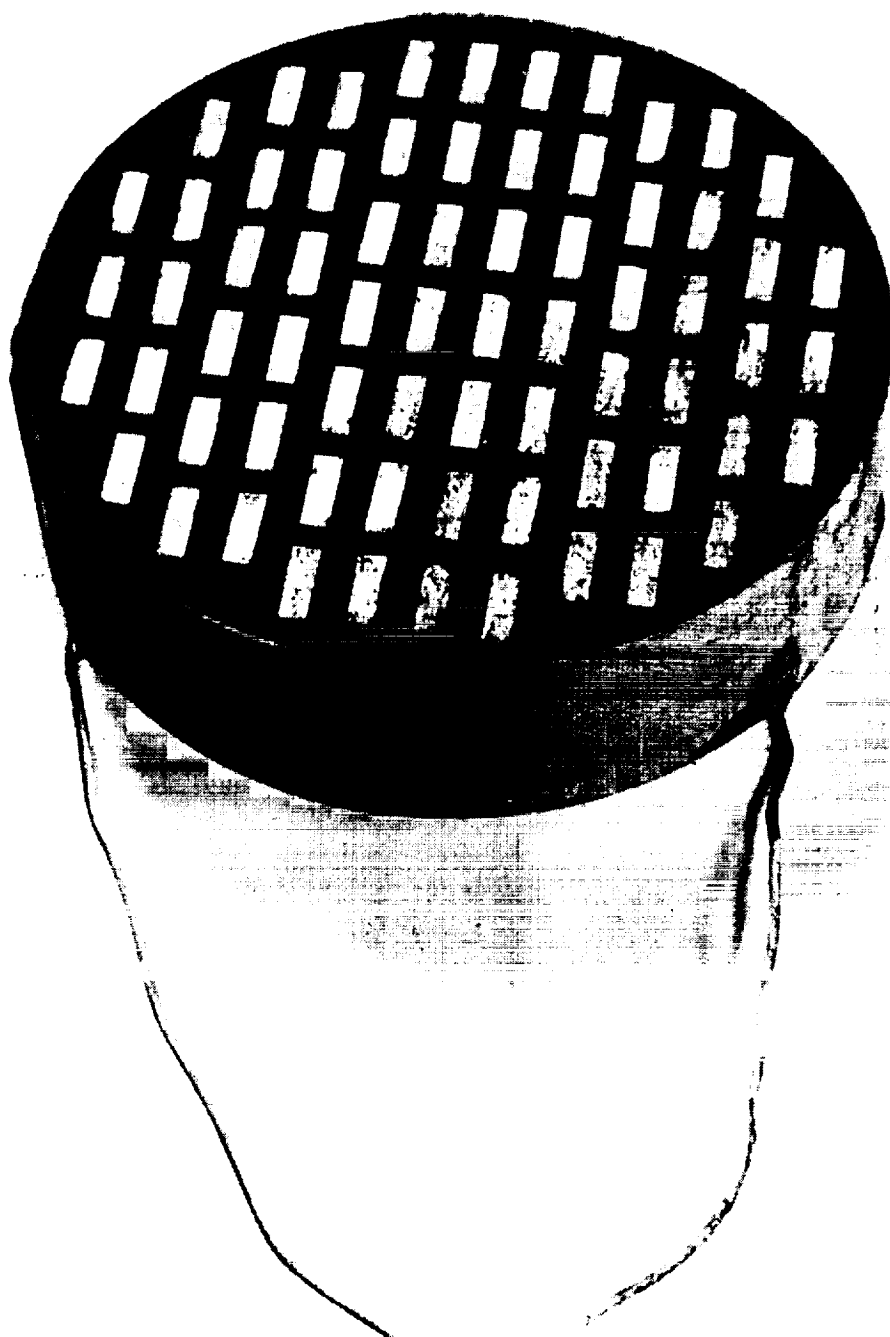
TWO-WATT SPECIAL APPLICATIONS TERRESTRIAL RTG **(BeCu Housing Design For 10,000 Psi External Pressure)**



BOL POWER OUTPUT = 5.0 W(e)
 THERMAL INVENTORY = 72 W(t)
 APPROXIMATE WEIGHT = 30 LBS.

Viewgraph 2

**SPECIAL APPLICATIONS MODULE
(HOT SIDE VIEW)**



Viewgraph 3

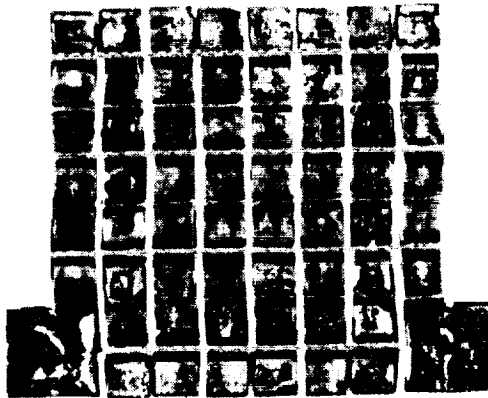
TWO-WATT SPECIAL APPLICATIONS RTG PERFORMANCE PREDICTION SUMMARY

	<u>WORST-CASE IN-AIR (BOL)</u>	<u>IN-WATER (BOL)</u>	<u>IN-WATER (10 YRS.)</u>
POWER OUTPUT (W(E))	3.95	4.92	3.37 (2.6 @ .99 TO 0.999 REL.)
FUEL INVENTORY (W(T))	70.8	70.8	65.4
T/E EFFICIENCY (%)	7.74	9.36	6.89
THERMAL EFFICIENCY (%)	73.2	75.1	74.8
SYSTEM EFFICIENCY (%)	5.58	6.94	5.15
HOT JUNCTION TEMPERATURE (°F)	814	676	641
COLD JUNCTION TEMPERATURE (°F)	214	50	50
AMBIENT TEMPERATURE (°F)	113	40	40

NOTES: (1) T/E ELEMENT DIMENSIONS: 0.450 IN. LG. x 0.103 IN. SQ.
(2) NUMBER T/E COUPLES: 68
(3) RTG FILL GAS: 100% XENON

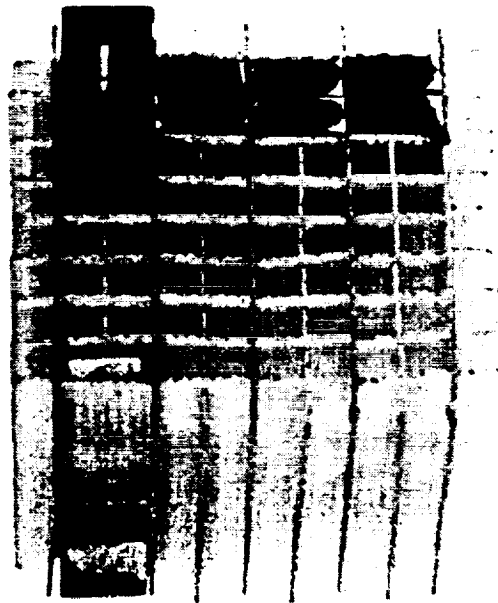
Viewgraph 4

**30-COUPLE (.067" SQ. ELEMENTS)
DEVELOPMENT MODULE QUADRANT
(COLD SIDE VIEW)**



Viewgraph 5

**30-COUPLE (.067" SQ. ELEMENTS)
DEVELOPMENT MODULE QUADRANT
(PARTIALLY "STUFFED" WITH COUPLES)**



Viewgraph 6

ORIGINAL PAGE IS
OF POOR QUALITY

SPECIAL APPLICATIONS PROGRAM

30-COUPLE MODULE PERFORMANCE

(ELEMENT SIZE = .067" SQ. X .483" LG.)

	<u>PREDICTION</u>	<u>QUADRANT #3</u> <u>(11/11/84)</u>
POWER OUTPUT (WATTS(e))	1.27	1.28
POWER INPUT (WATTS(t))	16.3 (1)	15.8 (2)
HOT JUNCTION (°F)	925	925 ⁽³⁾
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (VDC)	4.80	4.79
LOAD VOLTAGE (VDC)	2.40	2.41
INTERNAL RESISTANCE (Ω)	4.55 ⁽⁴⁾	4.48
EXTRANEIOUS RESISTANCE (%)	0	-1.5
THERMOELECTRIC EFFICIENCY (%)	9.3	-

(1) INCLUDES: $Q_{T/E}$ (14.1 W) + $Q_{SEPARATORS}$ (2.2 W)

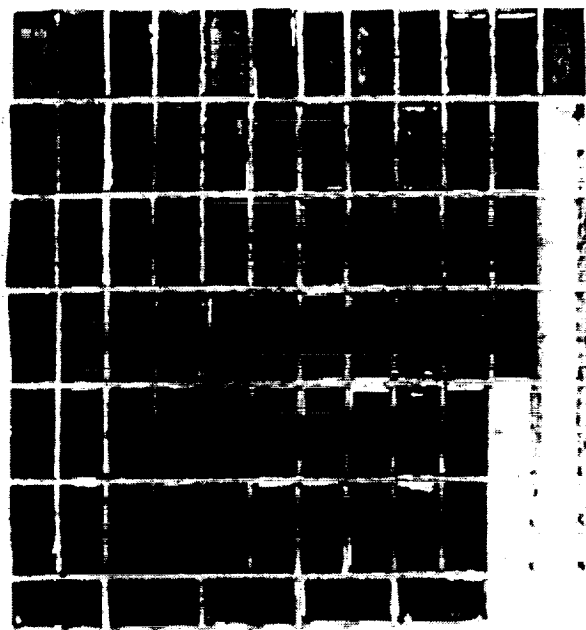
(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

(4) INCLUDES $R_{T/E}$ (4.38 Ω) + R_{STRAPS} (.07 Ω) + R_{LEADS} (.10 Ω).

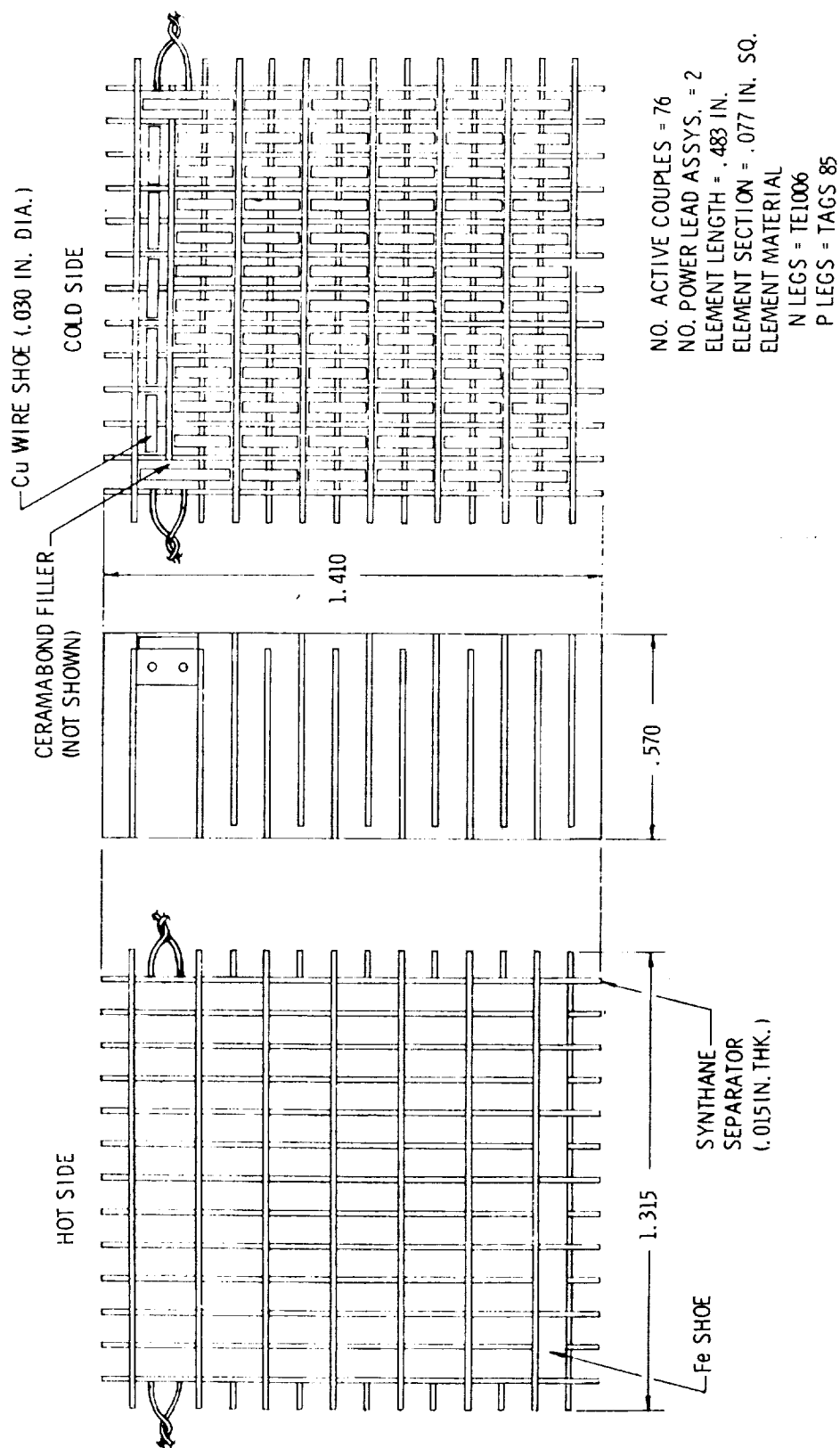
Viewgraph 7

SPECIAL APPLICATIONS TECHNOLOGY PARTIALLY ASSEMBLED MODULE ASSEMBLY



Viewgraph 8

SPECIAL APPLICATIONS PROGRAM 76-COUPLE DEVELOPMENT MODULE



Viewgraph 9

SPECIAL APPLICATIONS PROGRAM

76-COUPLE MODULE PERFORMANCE

(ELEMENT SIZE = .077" SQ. X .483" LG.)

	<u>PREDICTION</u>	<u>MODULE S/N 6 (11/3/84)</u>
POWER OUTPUT (W(e))	4.25	4.26
POWER INPUT (W(t))	56.1 ⁽¹⁾	56.2 ⁽²⁾
HOT JUNCTION (°F)	925	925 ⁽³⁾
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (V)	12.16	12.14
LOAD VOLTAGE (V)	6.10	6.10
INTERNAL RESISTANCE (Ω)	8.69 ⁽⁴⁾	8.65
EXTRANEIOUS RESISTANCE (%)	0	-0.5
THERMOELECTRIC EFFICIENCY (%)	9.3	—

(1) INCLUDES: $Q_{T/E}$ (47.3 W) + $Q_{SEPARATORS}$ (8.8 W).

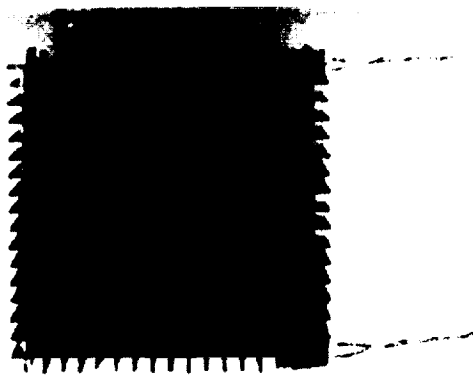
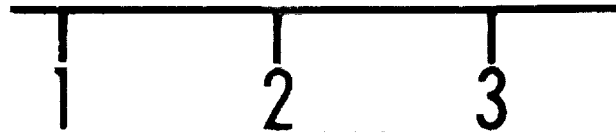
(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

(4) INCLUDES: $R_{T/E}$ (8.41 Ω) + R_{STRAPS} (.18 Ω) + R_{LEADS} (.10 Ω).

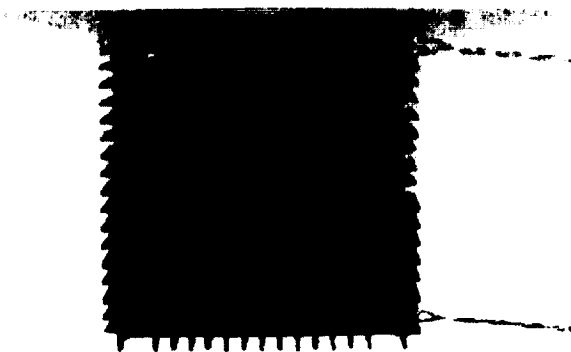
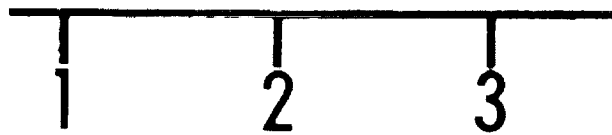
FIVE-WATT DEVELOPMENT THERMOELECTRIC MODULE

SCALE



COLD SIDE OF COMPLETED MODULE

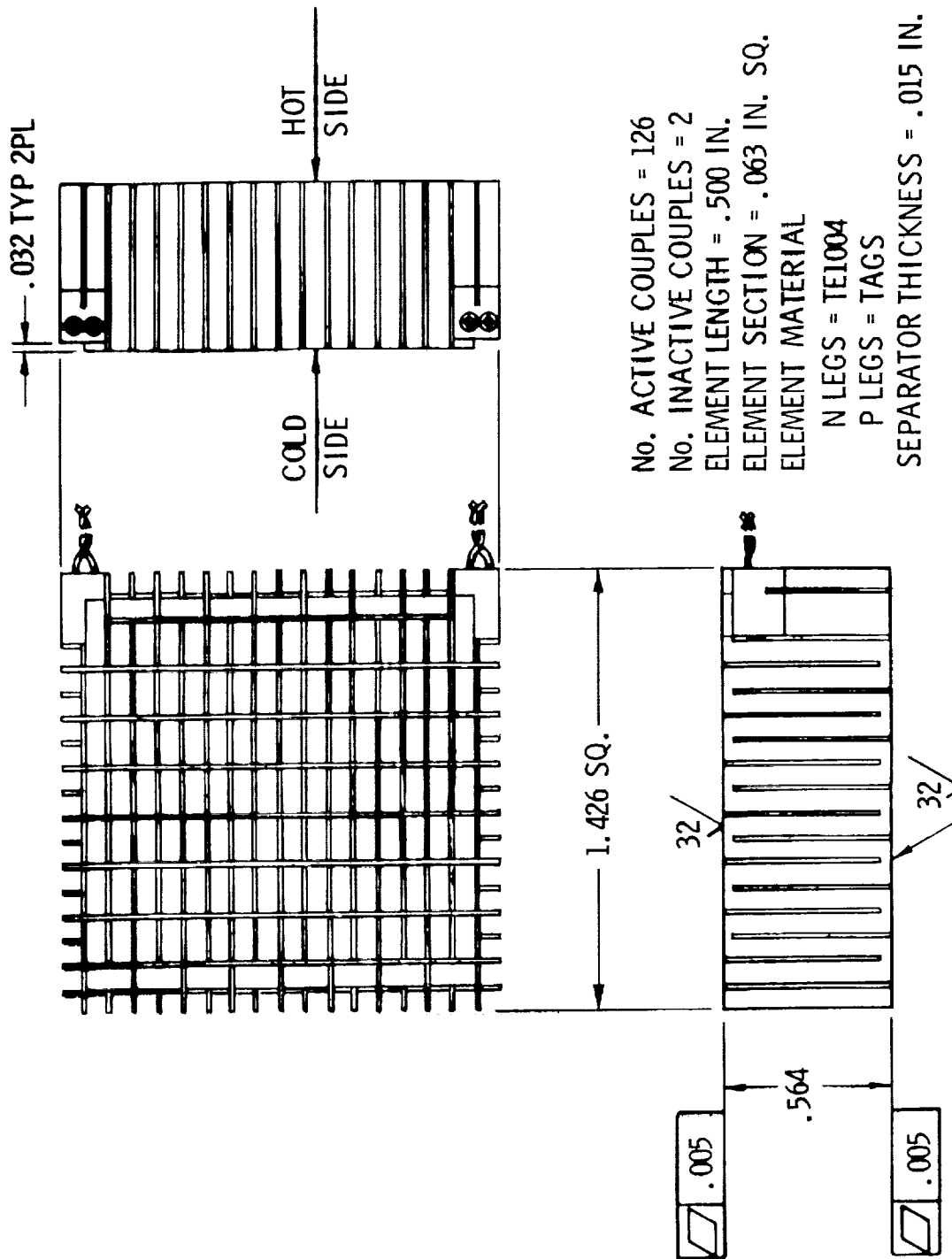
SCALE



HOT SIDE OF COMPLETED MODULE

Viewgraph 11

THERMOELECTRIC MODULE ARRANGEMENT



Viewgraph 12

SPECIAL APPLICATIONS PROGRAM

126-COUPLE MODULE PERFORMANCE

(ELEMENT SIZE = .061" X .466" LG.)

	<u>PREDICTION</u>	<u>MODULE S/N 5 (10/12/84)</u>
POWER OUTPUT (WATTS(e))	4.51	4.47
POWER INPUT (WATTS(t))	59.7 ⁽¹⁾	60.0 ⁽²⁾
HOT JUNCTION (°F)	925	925 ⁽³⁾
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (VDC)	19.27	19.30
LOAD VOLTAGE (VDC)	9.63	9.72
INTERNAL RESISTANCE (Ω)	20.56 ⁽⁴⁾	20.83
EXTRANEIOUS RESISTANCE (%)	0	1.3
THERMOELECTRIC EFFICIENCY (%)	9.2	—

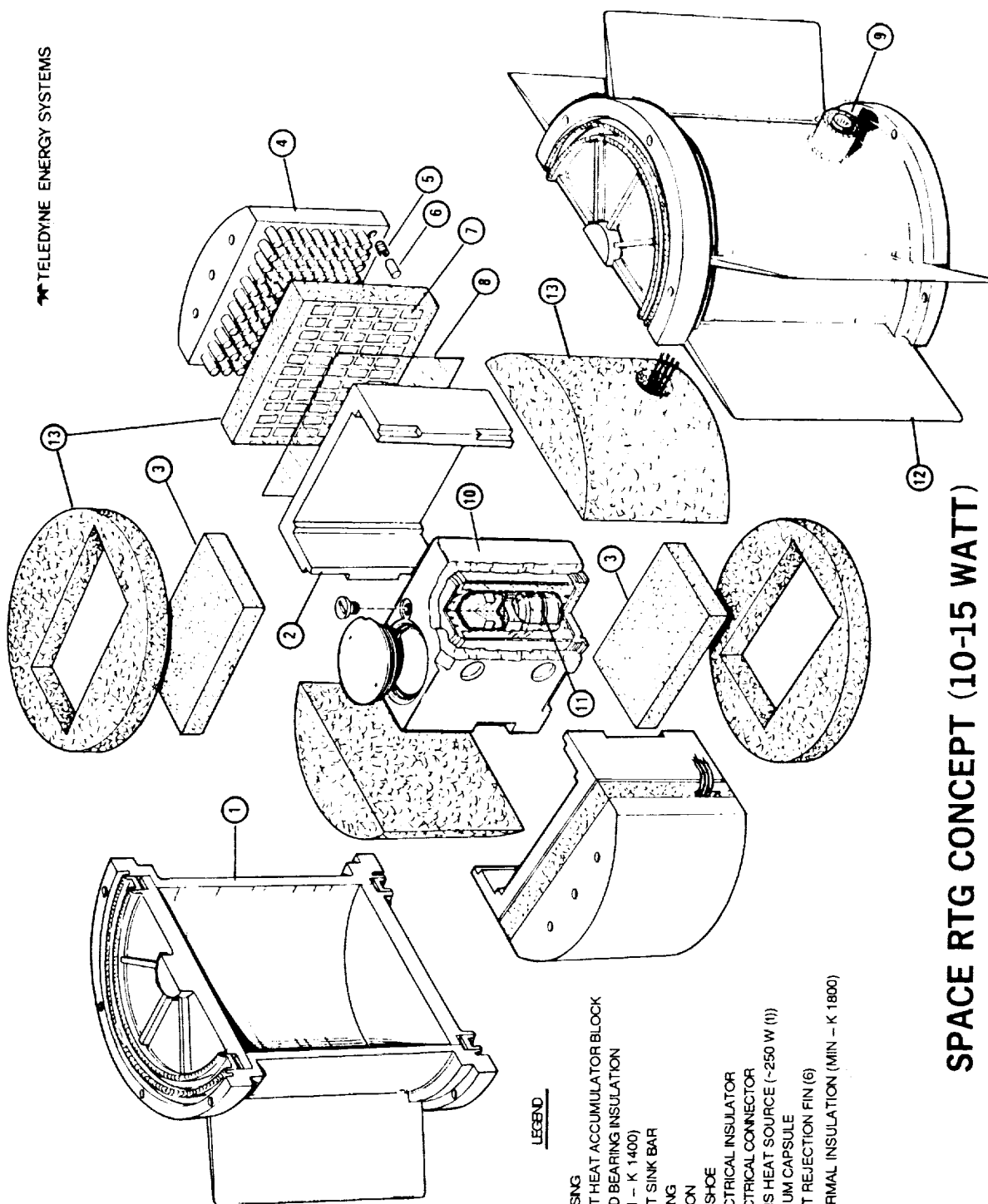
(1) INCLUDES: $Q_{T/E}$ (49.4 W) + $Q_{SEPARATORS}$ (8.2 W) +

$Q_{INERT COUPLES(5)}$ (2.1 W).

(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

(4) INCLUDES $R_{T/E}$ (20.40 Ω) + R_{STRAPS} (.06 Ω) + R_{LEADS} (.10 Ω).



LEGEND

- 1 HOUSING
- 2 SPLIT HEAT ACCUMULATOR BLOCK
(MIN - K 1400)
- 3 LOAD BEARING INSULATION
(MIN - K 1400)
- 4 HEAT SINK BAR
- 5 SPRING
- 6 PISTON
- 7 HOT SHOE
- 8 ELECTRICAL INSULATOR
- 9 ELECTRICAL CONNECTOR
- 10 GPHS HEAT SOURCE (~250 W (1))
- 11 IRIIDIUM CAPSULE
- 12 HEAT REJECTION FIN (6)
- 13 THERMAL INSULATION (MIN - K 1800)

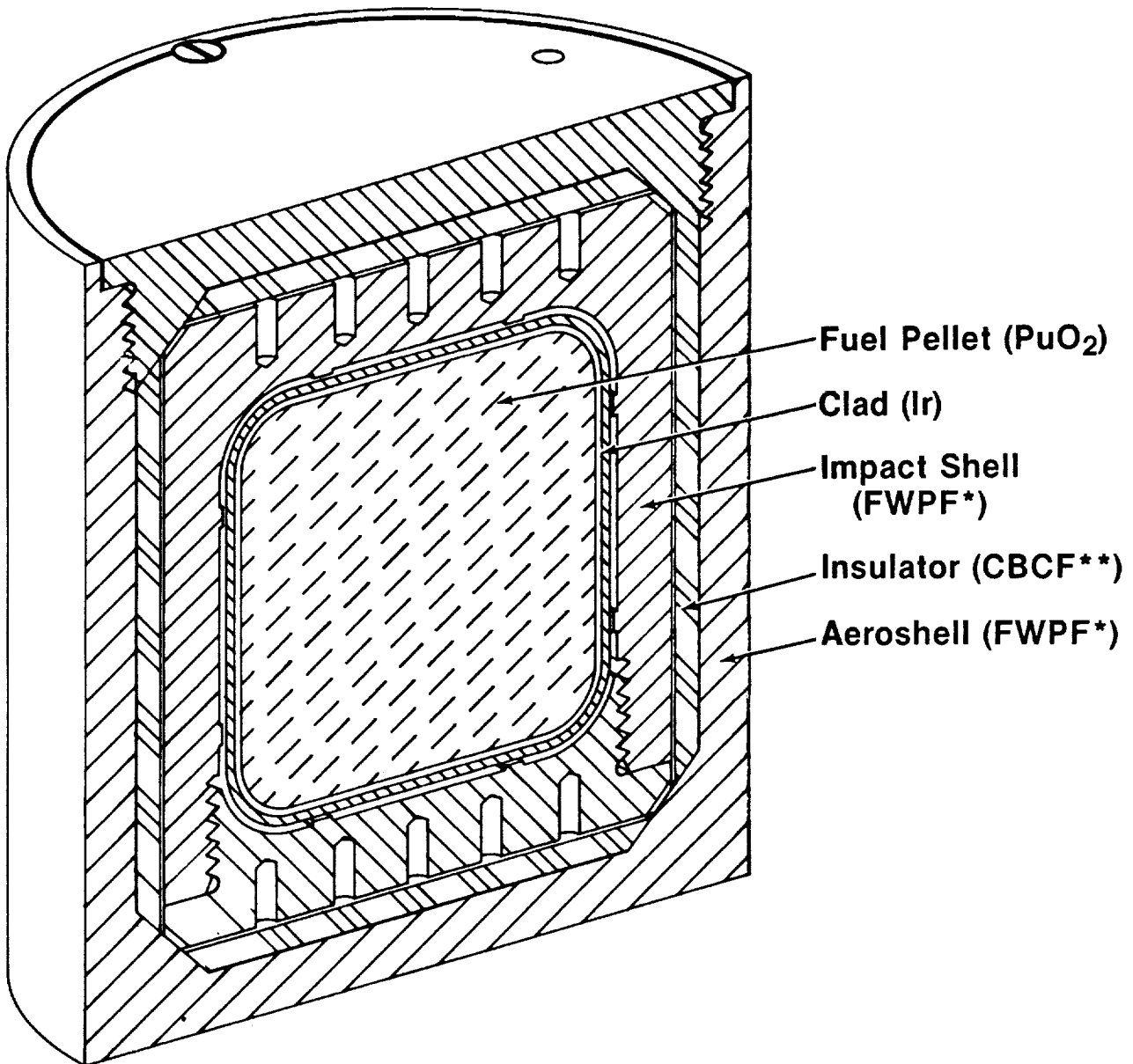
SPACE RTG CONCEPT (10-15 WATT)

Viewgraph 14

Session C, Submittal No. 6

Alfred Schock
Fairchild Space Company

RADIOISOTOPE HEAT SOURCE



*Fine-Weave Pierced Fabric)

**Carbon-Bonded Carbon Fibers

Technical drawing of a cylindrical container, showing a cross-section (top view) and an end view (bottom view).

Dimensions (Top View):

- Overall diameter: $\varnothing 1.550$
- Inner diameter: $\varnothing 1.173$
- Outer diameter: $\varnothing 1.715$
- Wall thickness: $.025$
- Inner radius: $.167$
- Outer radius: $.070$
- Inner height: 1.677
- Outer height: 1.217
- Overall height: 2.159
- Inner radius offset: $.156$
- Outer radius offset: $.156$
- Inner radius offset: 1.080

Dimensions (End View):

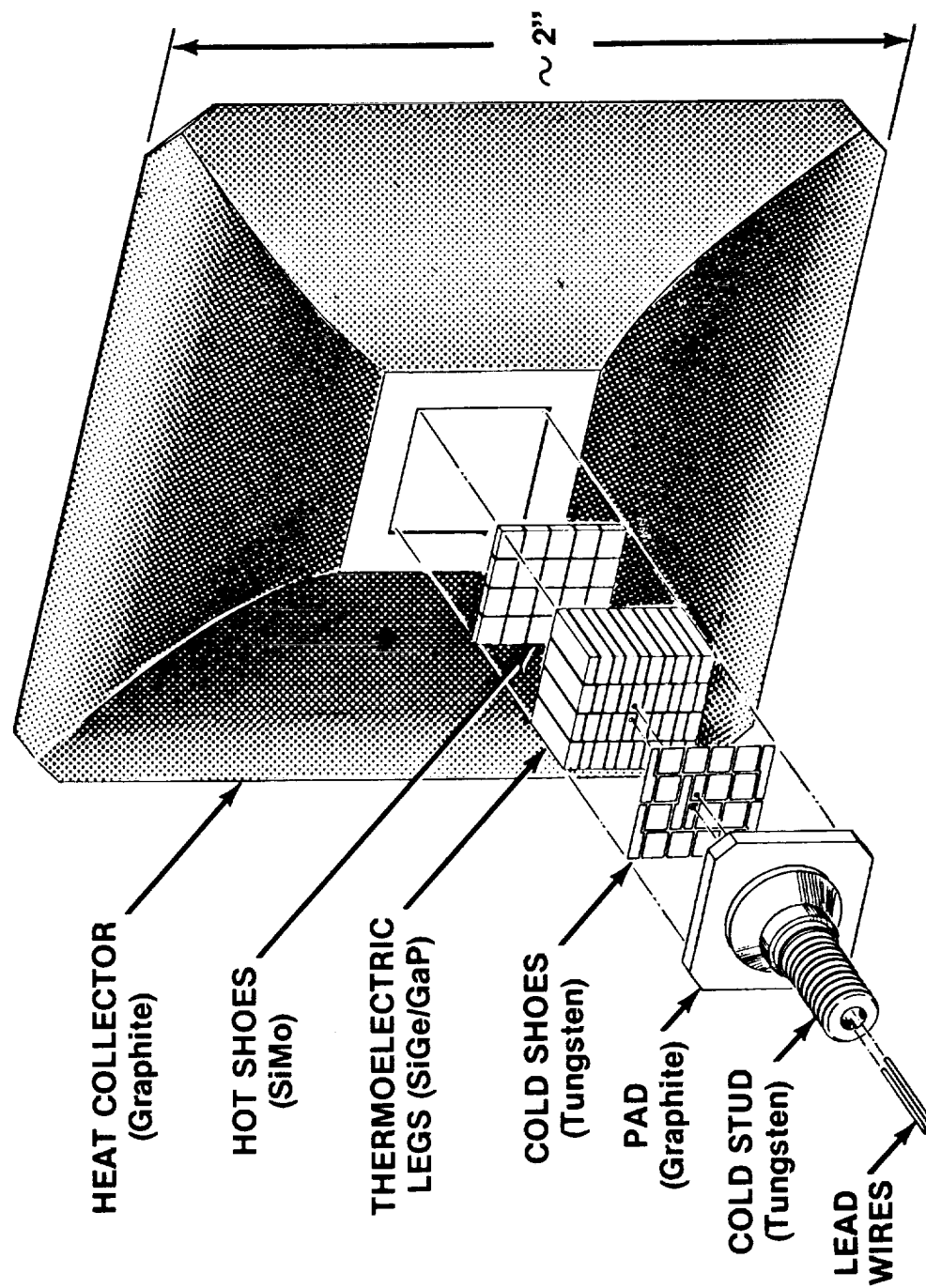
- Overall diameter: $\varnothing 2.032$
- Inner diameter: $\varnothing 1.083$

Material Labels:

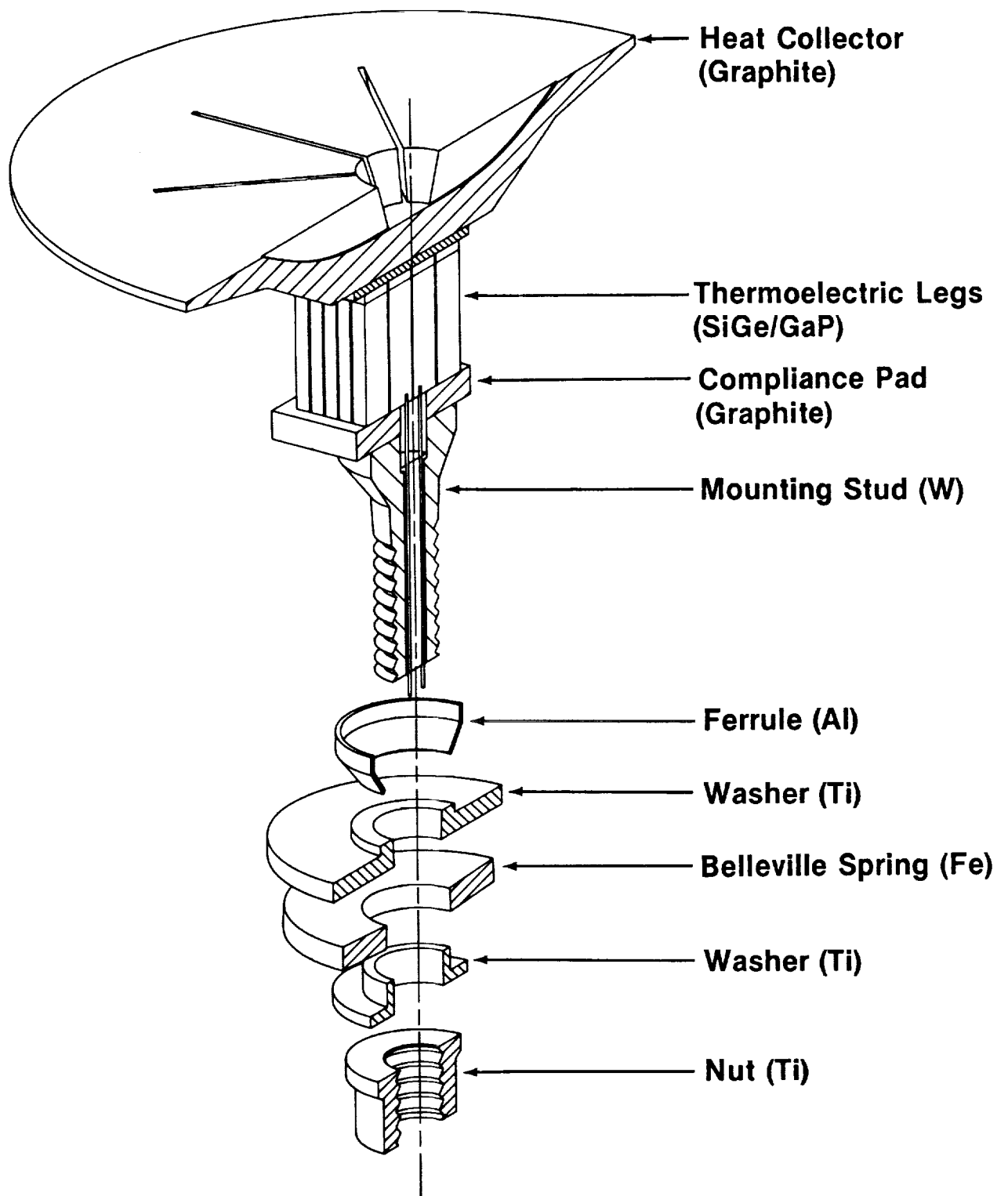
- Thermal Insulation CBCF (0.2)
- Impact Shell FWPF (1.95)
- Aero Shell FWPF (1.95)

Mass - 1.346 LB = 0.611 kg

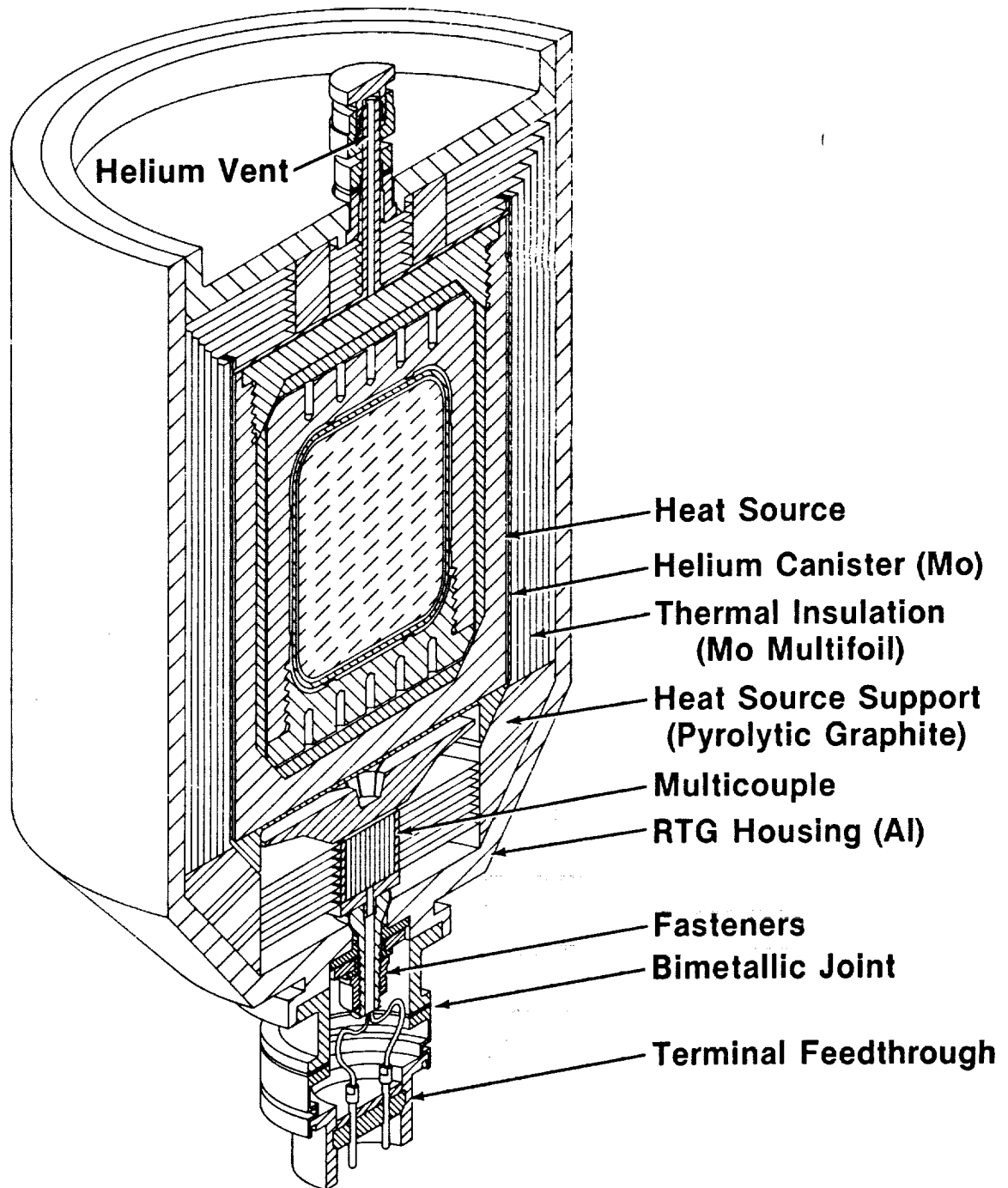
EXPLODED VIEW OF MULTICOUPLE (2.6 Watt, 3.5 Volt)



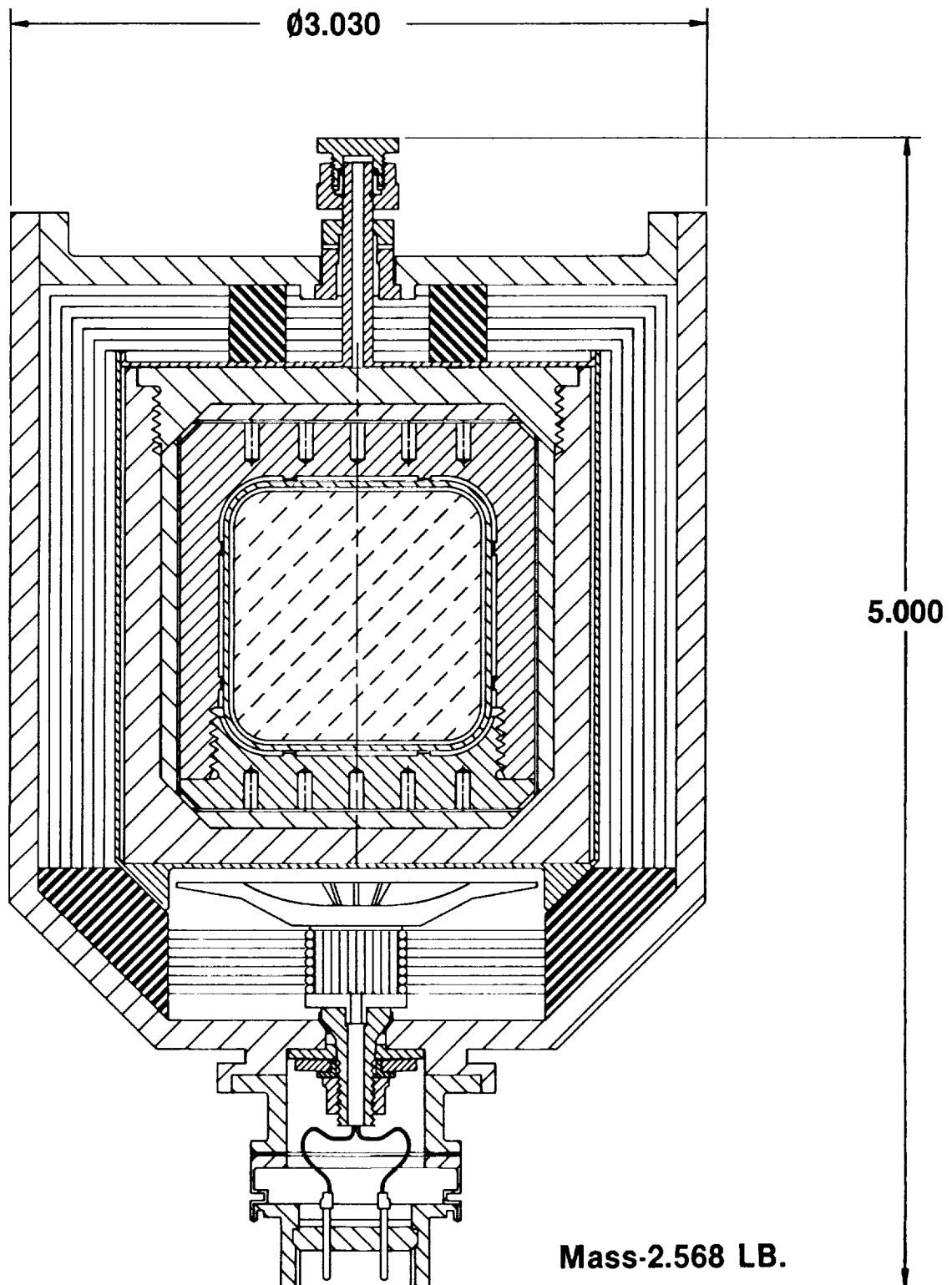
MULTICOUPLE AND FASTENERS



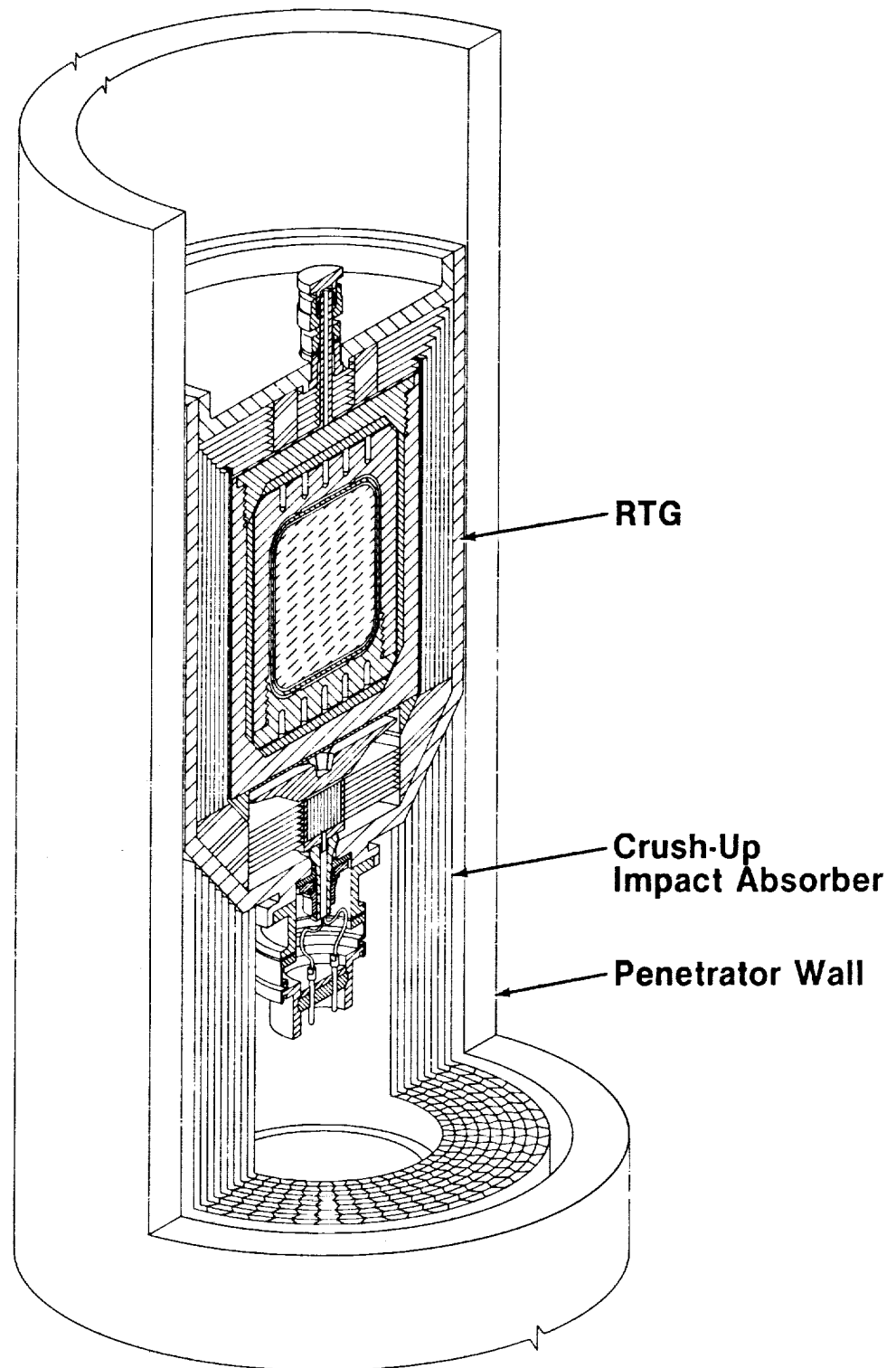
RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG)



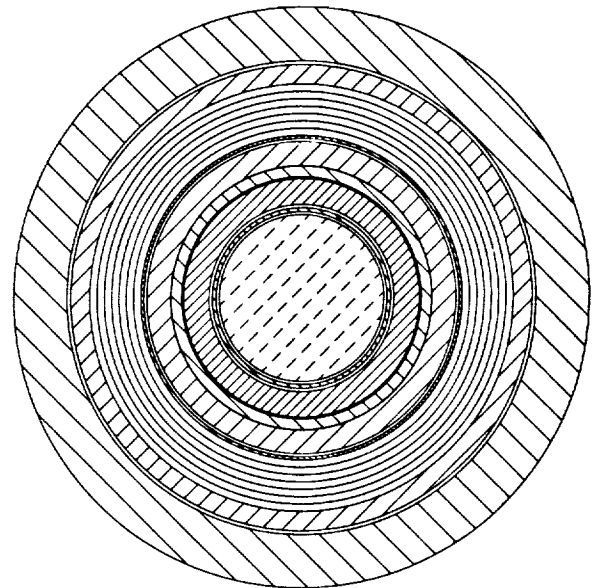
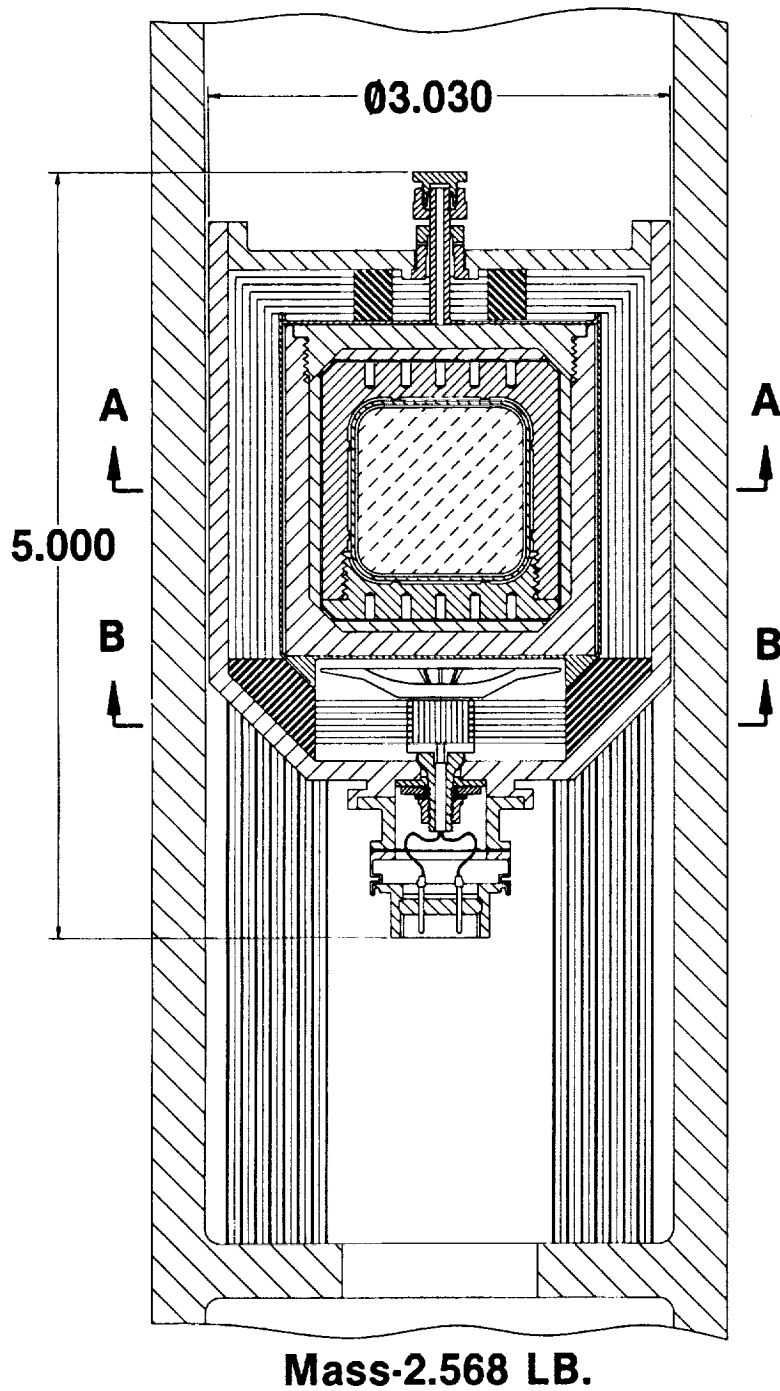
RTG CROSS-SECTIONS



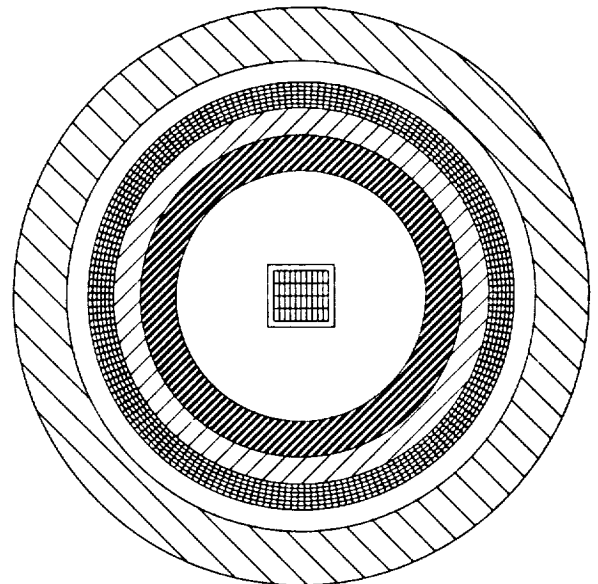
RTG IN PENETRATOR



RTG IN PENETRATOR CROSS-SECTIONS



SECTION A-A



SECTION B-B

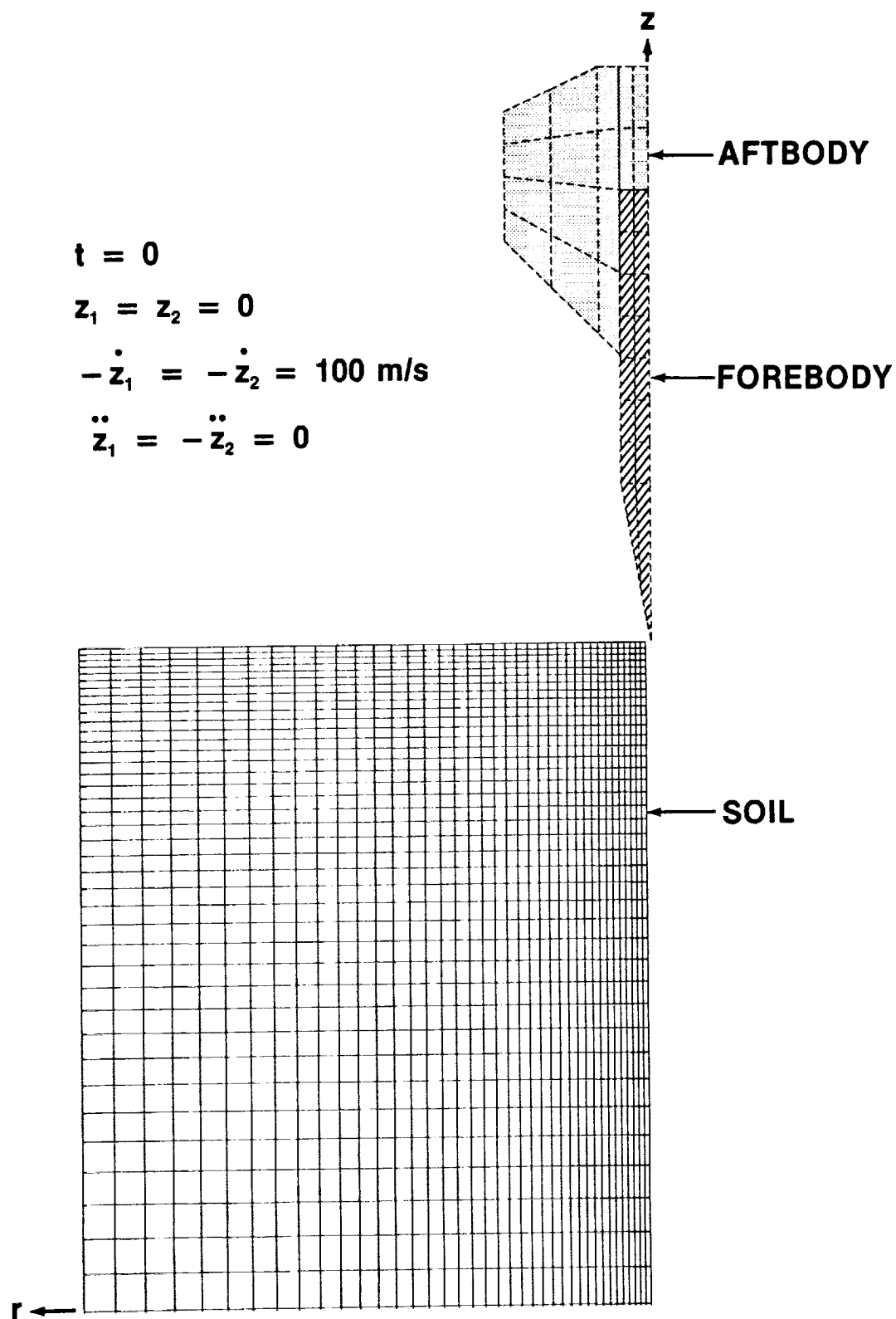
PENETRATOR DECELERATION, PAGE 1

$$t = 0$$

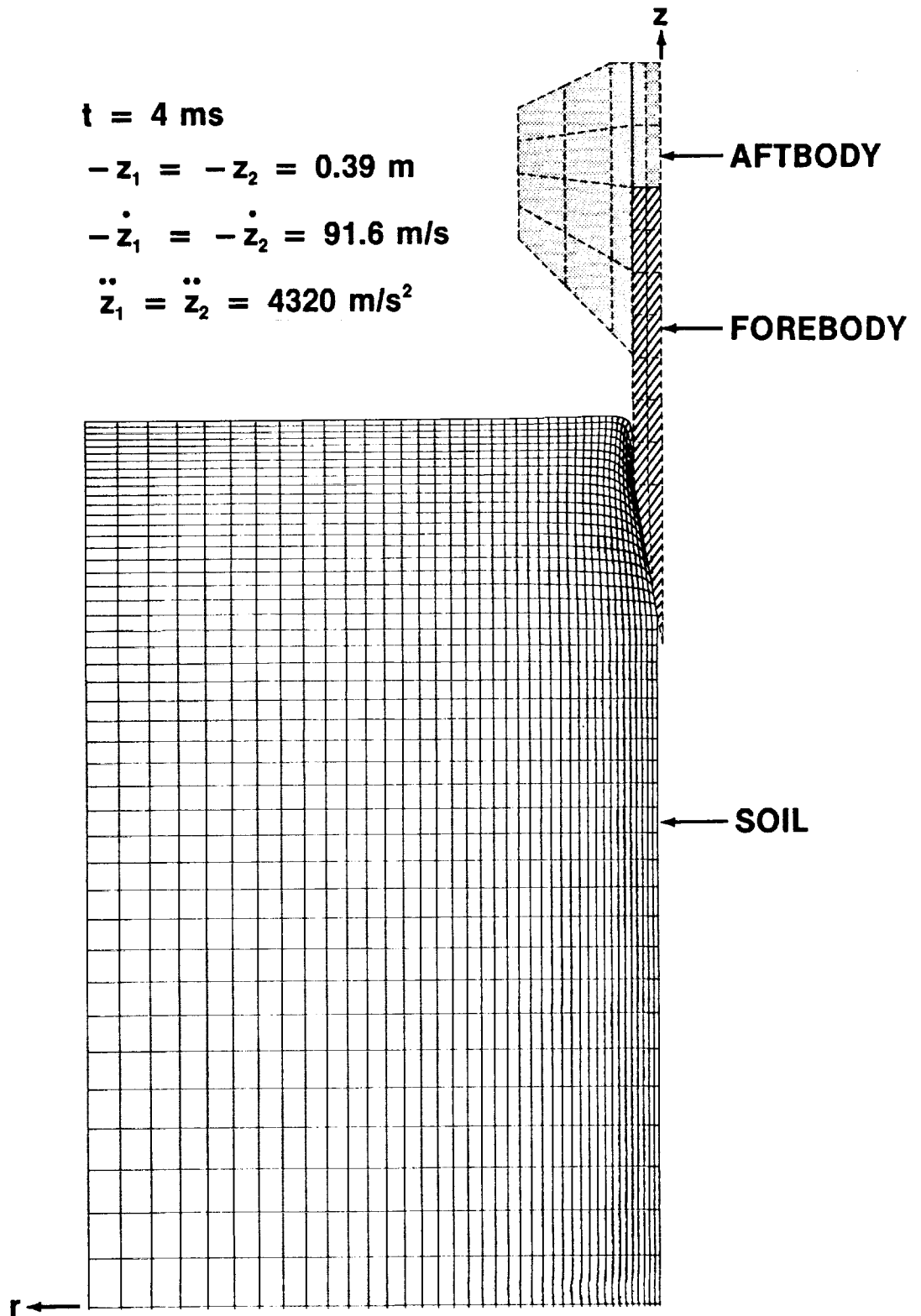
$$z_1 = z_2 = 0$$

$$-\dot{z}_1 = -\dot{z}_2 = 100 \text{ m/s}$$

$$\ddot{z}_1 = -\ddot{z}_2 = 0$$



PENETRATOR DECELERATION, PAGE 2



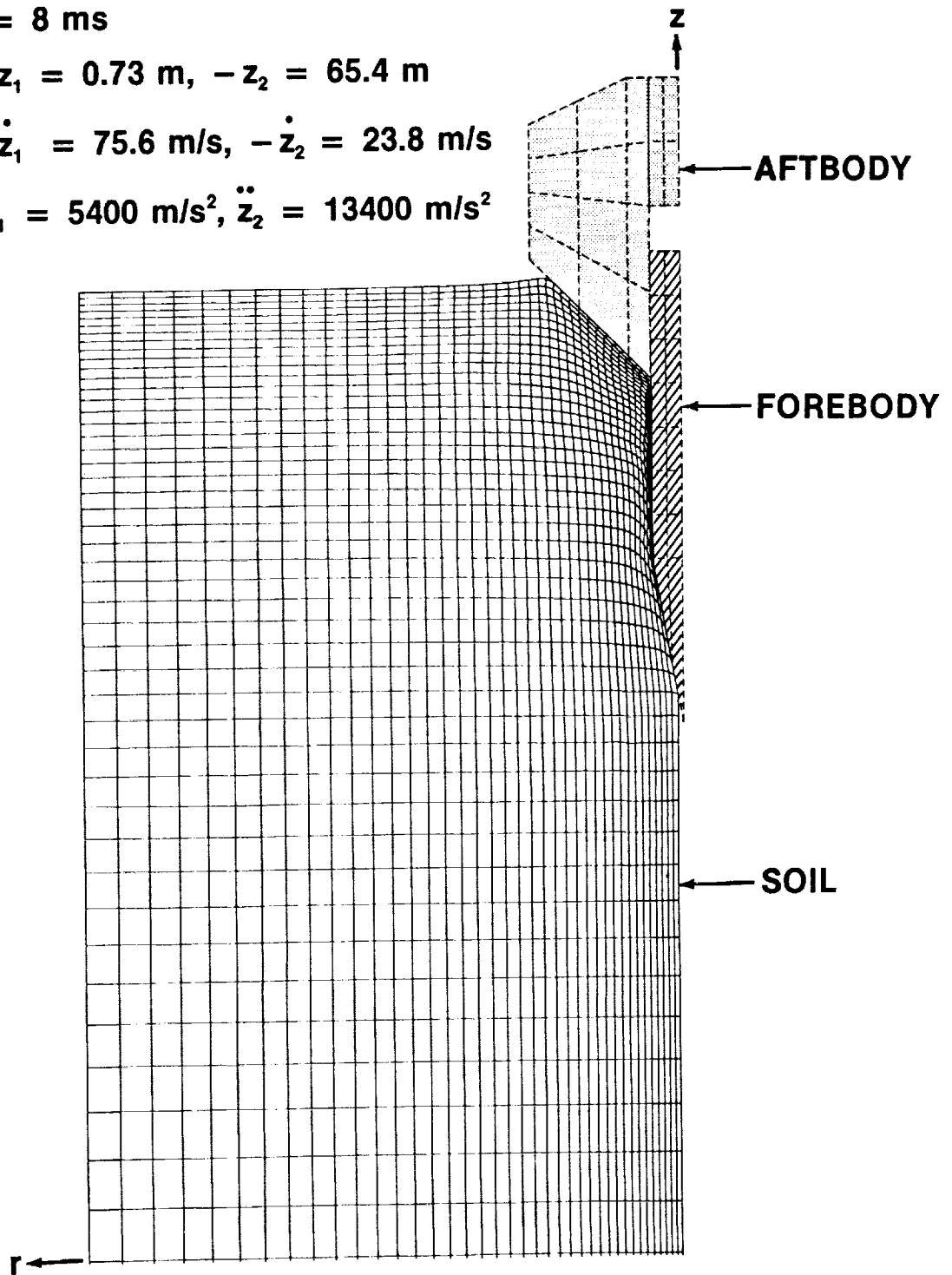
PENETRATOR DECELERATION, PAGE 3

$$t = 8 \text{ ms}$$

$$-z_1 = 0.73 \text{ m}, -z_2 = 65.4 \text{ m}$$

$$-\dot{z}_1 = 75.6 \text{ m/s}, -\dot{z}_2 = 23.8 \text{ m/s}$$

$$\ddot{z}_1 = 5400 \text{ m/s}^2, \ddot{z}_2 = 13400 \text{ m/s}^2$$



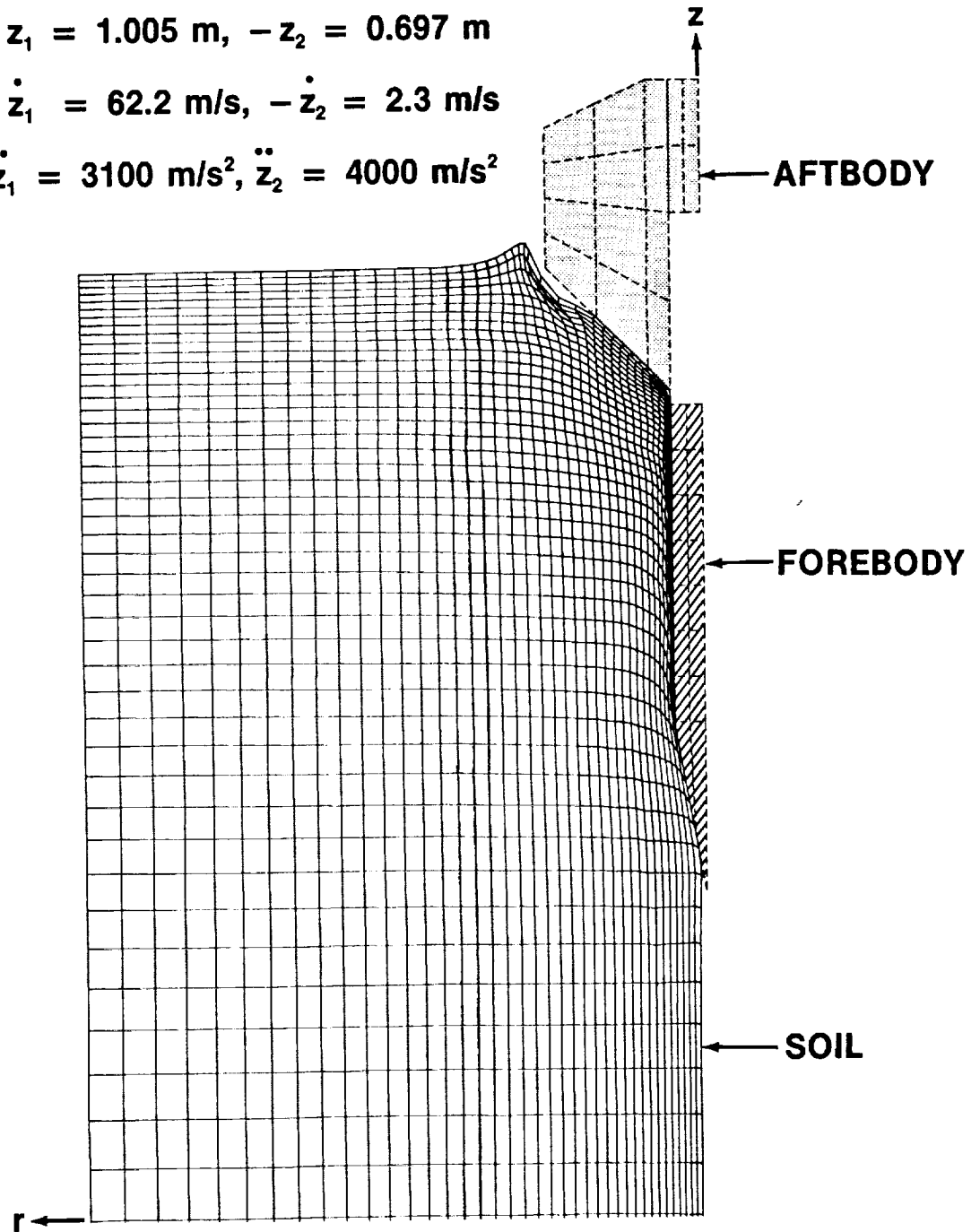
PENETRATOR DECELERATION, PAGE 4

$t = 12 \text{ ms}$

$-z_1 = 1.005 \text{ m}, -z_2 = 0.697 \text{ m}$

$-\dot{z}_1 = 62.2 \text{ m/s}, -\dot{z}_2 = 2.3 \text{ m/s}$

$\ddot{z}_1 = 3100 \text{ m/s}^2, \ddot{z}_2 = 4000 \text{ m/s}^2$



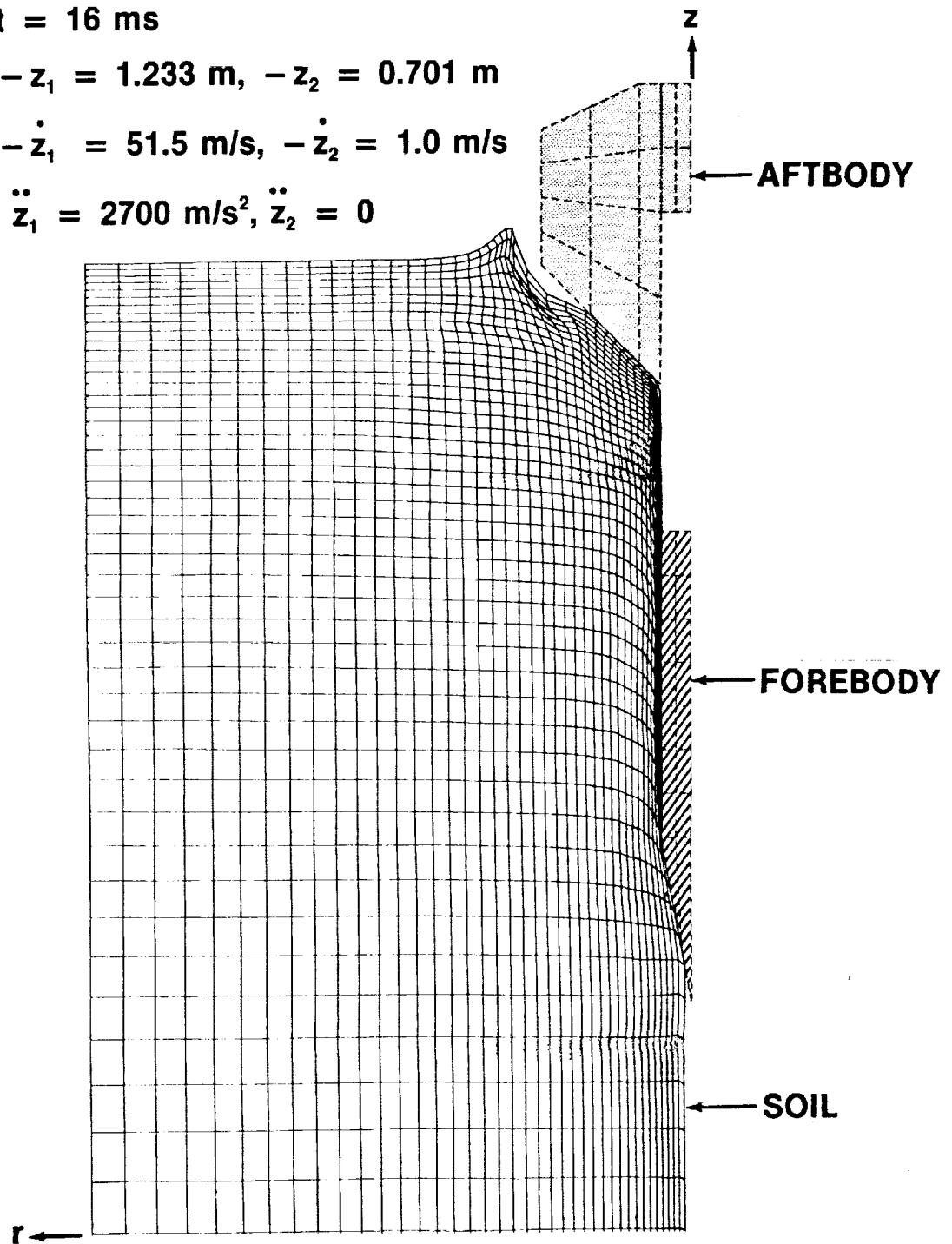
PENETRATOR DECELERATION, PAGE 5

$$t = 16 \text{ ms}$$

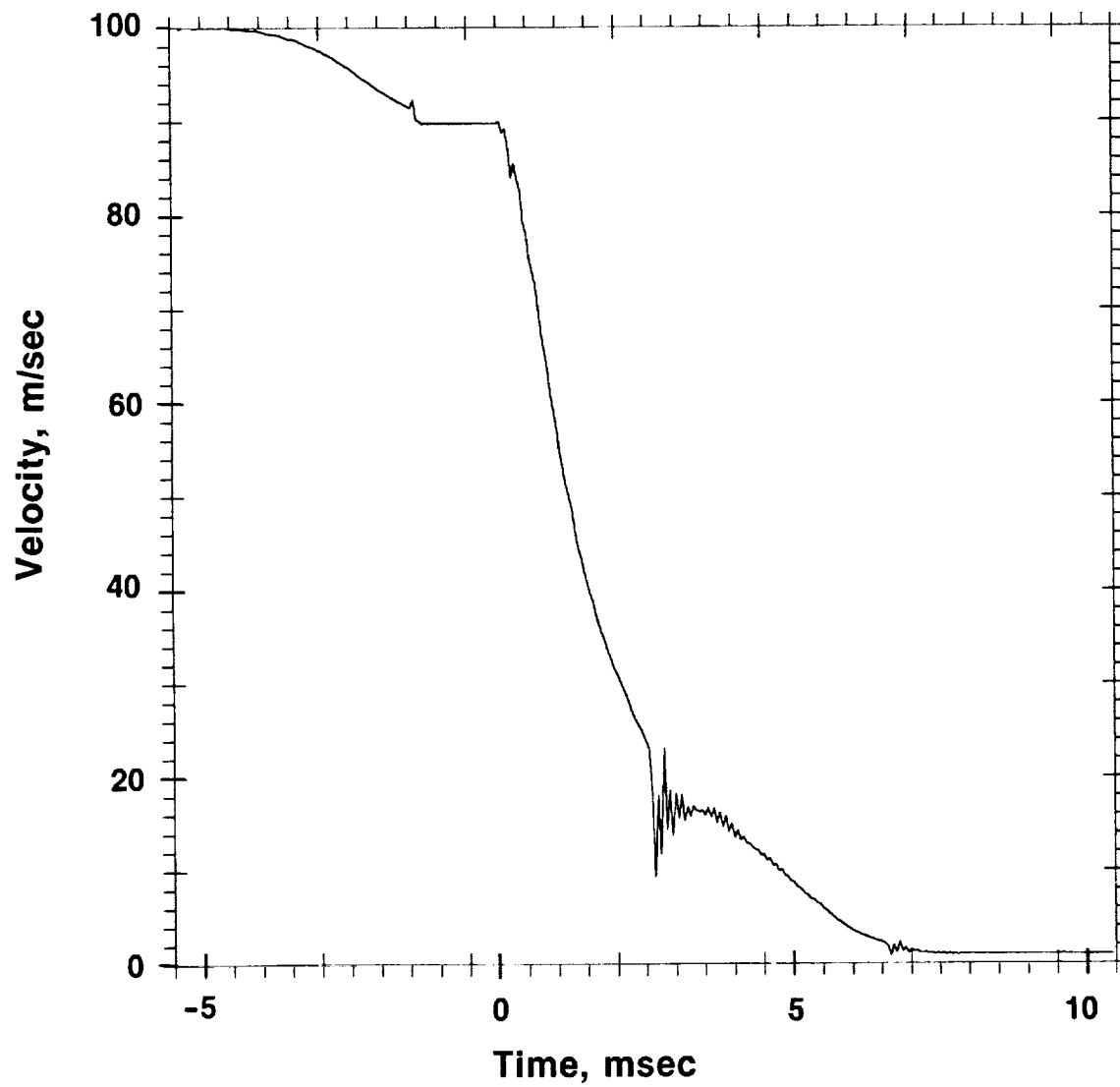
$$-z_1 = 1.233 \text{ m}, -z_2 = 0.701 \text{ m}$$

$$-\dot{z}_1 = 51.5 \text{ m/s}, -\dot{z}_2 = 1.0 \text{ m/s}$$

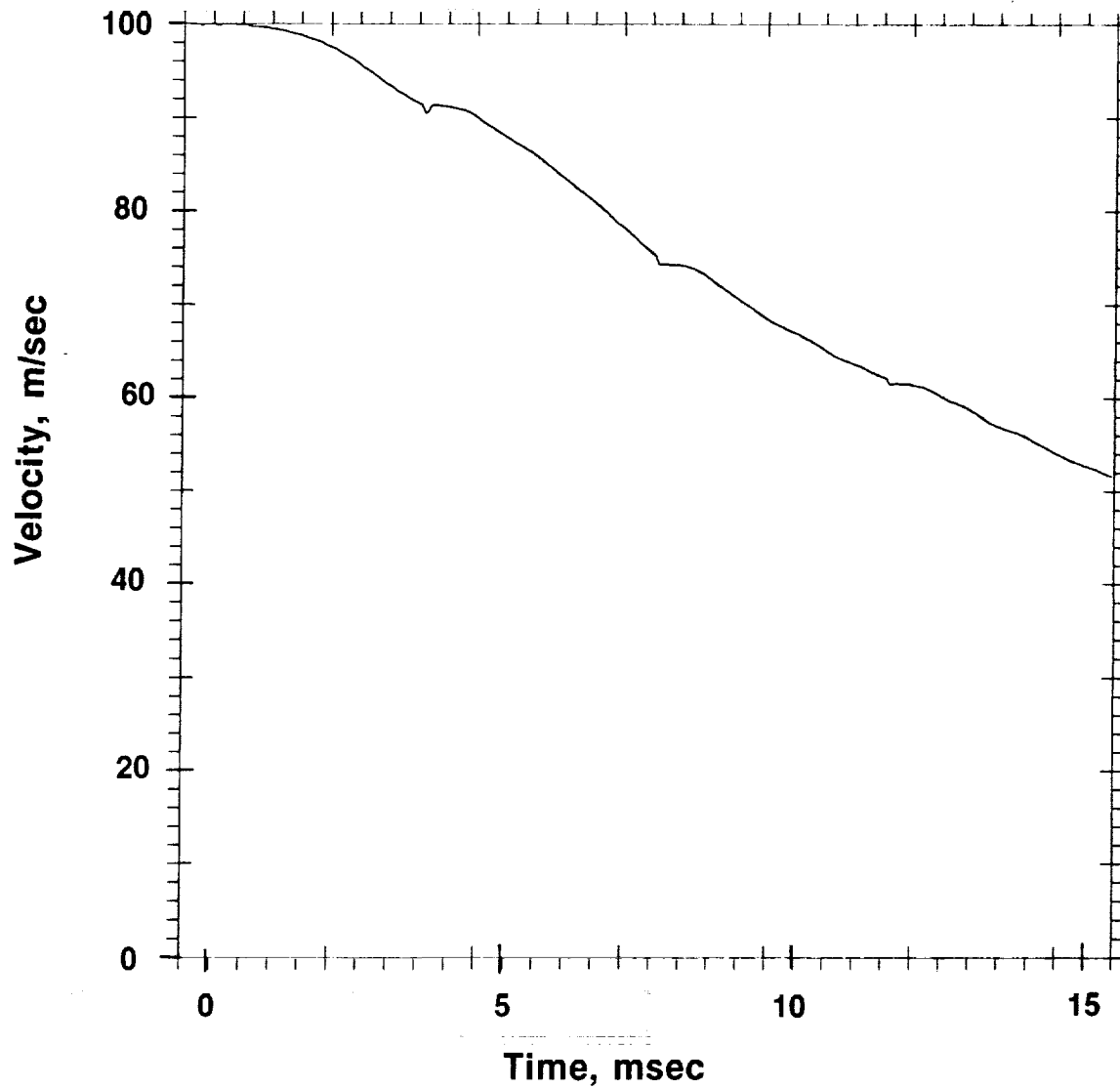
$$\ddot{z}_1 = 2700 \text{ m/s}^2, \ddot{z}_2 = 0$$



DECELERATION OF AFT BODY, VELOCITY VERSUS TIME AFTER IMPACT



DECELERATION OF FOREBODY, VELOCITY VERSUS TIME AFTER IMPACT



DECELERATION OF FOREBODY, VELOCITY VERSUS PENETRATION

